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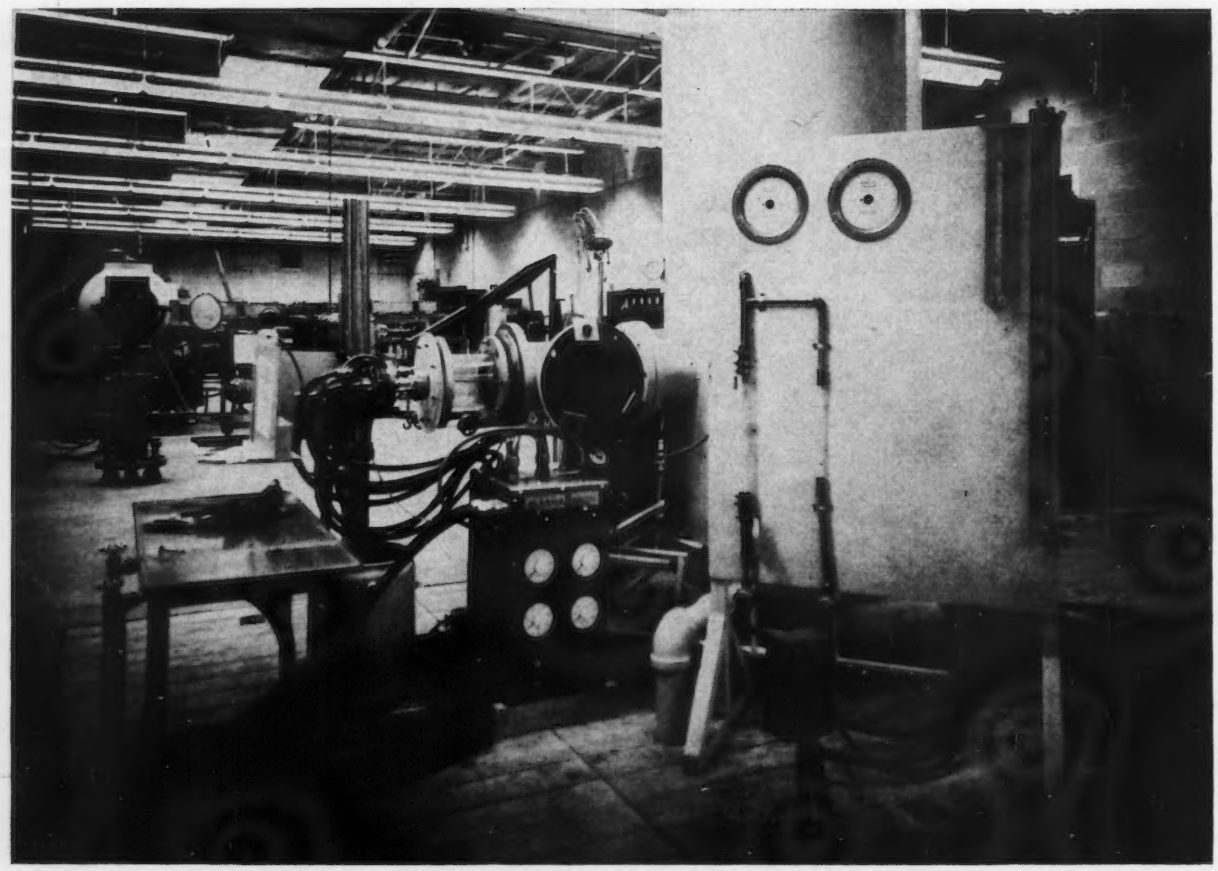
U.T.I.A. PLASMA TUNNEL



General view of main laboratory in the new Institute of Aerophysics Building. Pilot model of low density plasma tunnel in foreground (See page 206).

EDITORIAL

THE BLACK BOX

With the ever-increasing importance of electronics in aeronautical engineering it is suggested that the aeronautical engineer should screw up his courage and, metaphorically, take the lid off the "black box".

The black box has become a symbol, an iron curtain beyond which the aeronautical engineer has no right and, it must be confessed, little inclination to enquire. To him it is a device magically producing a result; how it does so is no concern of his. He is wont to study a system, say the controls of a missile, through its aerodynamic, mechanical, metallurgical and even electrical intricacies until, sooner or later, the black box in all its mystery bars further investigation. Provided it produces the desired result he accepts it as something that he is not meant to understand.

Occasionally, in a half-hearted way, he may ask the electronics engineer to explain the thing to him. He is usually rewarded by a dissertation in the broadest generalities, because the electronics engineer does not appreciate his pupil's abysmal ignorance of the first principles of the subject and, finding that he is completely out of his depth in the engineering stages, reverts to explaining what it does – which the aeronautical engineer probably knew in the first place. And so the barrier remains and is strengthened. The electronics engineer thinks that the aeronautical engineer is not really interested, and the aeronautical engineer regards electronics as a lot of mumbo-jumbo that even the experts cannot explain intelligibly.

Because he is a human being himself (and not a black box) the aeronautical engineer feels that he can understand the working of a pilot but, because he is an engineer and not a doctor, he ought to know much more about the workings of a black box – not only in what it does but in how it does it. It is surely the duty of the Institute to try to break down the symbolism of the black box, to interest the aeronautical engineer in electronic engineering and, if necessary, to take him back to first principles and lead him gently forward from there. He will not become an electronics engineer by this method but he may develop an intelligent interest, which would be helpful to himself and to the electronic engineers with whom he has to work. A little knowledge may be a dangerous thing but, recognized as such, it is much better than no knowledge at all.

Perhaps the Branch programmes should include more papers on elementary electronics. Perhaps the Branches might arrange short familiarization courses on electronics for their members, at local schools. Perhaps members of the Institute who are electronic engineers would write "instructional" papers for the Journal; such papers should not be textbook material but broad summaries, sufficient to give the reader a working knowledge of the subject rather than an authoritative grasp of it. And perhaps, most important of all, members who now boast that they know nothing about electronics should put their pride in their pockets and try to learn something. It is strange that there is quite a different attitude towards metallurgy, which is, in its way, quite as specialized a field as electronic engineering.

And finally the Institute should encourage the electronics engineer to present and submit for publication in the Journal advanced papers on his subject – papers such as he would write for the electronic fraternity. Perhaps only a small number of the Institute's members would understand them; but the same could be said of some of the papers on aerodynamics, structures and propulsion, and nobody would question the suitability of these subjects in an aeronautical publication. Let aviation electronics take its place beside them. The Institute should cover every phase of aeronautical technical activity, with papers of the highest possible quality in all related fields – and, it may be added, for all levels of readership.

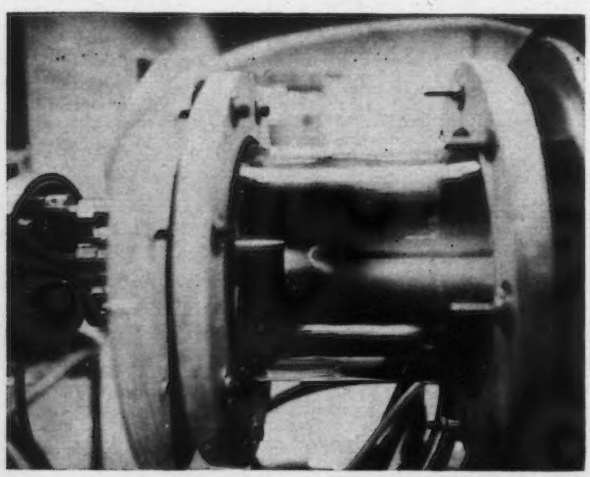
The mystery of the black box has no place in an engineering society. Let us take the lid off it, so that its insides will be recognized and developed as an integral part of the vehicle, in which all members of the Institute have individual but related interests.

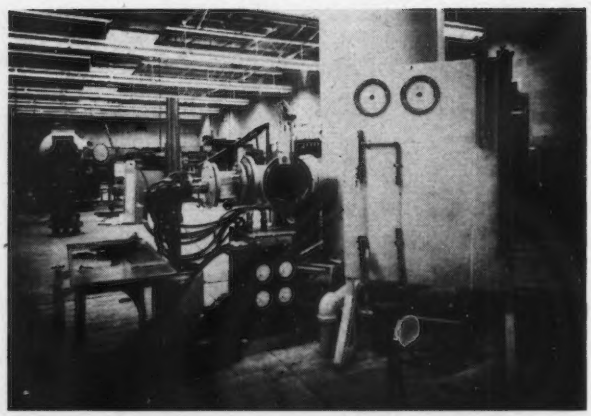
PLASMA TUNNEL

THE photographs illustrate a pilot facility used in the design study of a low density plasma tunnel to be constructed at the Institute of Aerophysics, University of Toronto, in the near future.

The tunnel working fluid (in this case argon) is passed through a 25 kw dc arc. Enough energy is added to achieve stagnation temperatures of over 10,000°K. Water cooled tungsten electrodes are used and the power and cooling water connections to the arc chamber are seen in the photograph. The argon is now a partially ionized plasma and is expanded from 30 mm Hg pressure in the arc chamber to about 2 mm Hg pressure in a simple conical nozzle. The nozzle survives because the plasma, which is hot enough to vaporize any solid, is separated from the nozzle wall by a cooler outer sheath of gas as a result of magnetogasdynamic forces.

The nozzle discharges at about Mach 2.5 into a pyrex cylinder, where the flow is being studied spectroscopically and with cooled probes for energy and velocity distributions. In the photograph, a blunt ceramic body





indicates, during its brief life, the supersonic nature of the plasma and the usual close approach to stagnation temperature behind the bow shock wave (suggested by the increased luminosity in that region).

In the proposed tunnel, this flow will be expanded much further to 10^{-5} atmospheres or less. This has necessitated the design of an unconventional cryogenic pumping system in which the working fluid from the test section is frozen out on a heat exchange surface maintained at 30°K (−243°C).

Studies in the whole field of low density, high energy are a natural outgrowth of the present low density research program at the Institute. This field is of interest in atmospheric entry and sustained flight at great altitudes, in proposed plasma and ionic space propulsion systems, and in basic studies of the phenomena pertaining to the extension of the kinetic theory of plasmas.

The research program is being conducted under the direction of Dr. G. N. Patterson by Dr. J. H. de Leeuw, Dr. I. I. Glass, Mr. J. B. French and Mr. E. P. Muntz, with the financial support of the Defence Research Board of Canada and the Air Force Office of Scientific Research (USAF).

ESTIMATION OF AIRCRAFT FATIGUE LIFE†

by E. Aubrey and B. Venkatraman*

Canadair Limited

SUMMARY

This paper presents a simplified method of calculating the life expectancy of an aircraft. Firstly, using the analyses of Chilver[1] and Harper[2] as bases, a relatively simple formula is derived for the damage per mile. Then, the fundamental assumptions made in estimating fatigue damage are discussed and equations for estimating aircraft fatigue life are presented. Direct reading charts for damage per mile and altitude corrections are given. The use of these is illustrated by calculating the fatigue life of a typical aircraft.

INTRODUCTION

In any discussion of aircraft life, the predominant considerations are those of designing for "safe-life" or "fail-safe". While the latter design criterion appears to be the more reasonable one for a successful and safe aircraft, it is still important to know the life of an aircraft from the economical viewpoint. The term "safe-life" implies that the aircraft will be safe for a specific length of time. Moreover, when "safe-life" is estimated, a prior consideration of fatigue failures due to badly fitted bolts, fretting of two parts rubbing together, and other such causes is not possible. Hence, the use of the term "safe-life" is highly questionable and in the authors' opinion it appears preferable to replace the term "safe-life" by "economical life".

From the preceding discussion it follows that an aircraft may be analyzed for economical life on the basis of failure solely due to fatigue. In such a procedure it must be ensured that the life of an aircraft is such that it will remain in service for a reasonable period before the appearance of fatigue cracks. When cracks and/or failures occur, then a "fail-safe" or "fail-slow" design will probably eliminate catastrophic failure. The design criterion is therefore one of "fail-slow", but at the same time a reasonable estimate of an economical crack-free life of the aircraft may be obtained.

A simple method of evaluating the "economical life" of an aircraft is presented in this paper. The method is based on the cumulative damage theory as presented by Miner[3]. Among others, Chilver[1] and Harper[2] have also based their methods of life prediction on the cumulative damage theory. An extension and simplification of these latter methods of analysis is presented here. The solution for the damage per mile has been obtained in a manner similar to that of Eggwertz[4] who used a more general representation of the S-N data. Further, it is shown that a given flight mission can be divided in such a manner as to considerably simplify the analysis of the total damage to the aircraft. Basically, under certain conditions usual in flight practice, the evaluation of damage per flight reduces to the use of a few direct reading charts. The estimation of the "economical life" then follows very simply as illustrated by the example.

LIST OF SYMBOLS

A, B	gust data constants
a, b	S-N curve ratios (constants)
C_h	altitude correction factor
D	horizontal distance – miles
c, d	S-N curve constants
E	frequency of gust or stress cycles
F'	gust alleviating factor
F_n	number of flight missions to failure
f	percentage alternating stress
f_c, f_g, f_s	correction factors
H, h	altitude – feet
L	lift – lb
ΔL	change in lift – lb
L_e, L_{ec}	aircraft life, theoretical and actual – hours
m	miles per gust
N	number of stress cycles to failure
n	ultimate load factor
Δn	load increment factor
S	wing area – feet^2
t	time – hours
V	aircraft speed – mph
v_h	gust velocity – fps
W	aircraft weight – lb
α	slope of lift curve per radian
σ_a	alternating stress – psi
σ_h	air density ratio
σ_u	ultimate stress – psi
Δ	damages
ρ	mass density of air – lb sec^2/in^4

†Received 12th November, 1958.
*Structures Development Section.

GUST-STRESS TRANSFORMATION

When an aircraft is on a level flight in still air, its wings and tailplanes are subjected to a steady load which corresponds to the total weight of the aircraft. If the aircraft is disturbed from this condition by a vertical gust, the consequent difference between the lift and the weight contributes an acceleration normal to the flight path of the aircraft. In civil aircraft under cruising conditions, the majority of superimposed loads are due to up and down gusts of varying intensity. For fatigue damage calculations, it is assumed that an up gust is always followed by a down gust of equal intensity. Thus, in addition to the steady load due to the weight of the aircraft, an alternating load cycle is superimposed on the wings.

In conditions of level flight, the stress in the various parts of the structure is usually called the 1g stress. In such flight then, assuming the wing supports the whole aircraft weight, the stress may be formally written as

$$\text{Stress} = \frac{L}{W} \times 1g$$

$$= 1g \tag{1}$$

The additional stresses imposed by gusts of varying intensity may then be considered as a factor or percentage of 1g.

When an aircraft flying at an altitude of h ft and an equivalent airspeed V_{eh} mph encounters a sharp edge gust of v_h fps, the change in lift on the wing is given by

$$\Delta L = \frac{1}{2} F' a S \rho v_h V_{eh} (22/15) \tag{2}$$

Here, F' is the gust alleviating factor, a is the slope of the lift curve per radian, S is the wing area in sq ft, ρ is the mass density of air and the factor 22/15 accounts for the transformation from mph to fps. Thus, the acceleration factor or the load increment factor is defined as

$$\Delta n = \frac{\Delta L}{W}$$

$$= \frac{F' a S \rho}{2W} v_h V_{eh} (22/15) \tag{3}$$

It follows then that the total stress imposed on the wing is

$$\text{Stress} = (1 \pm \Delta n) \times 1g \tag{4}$$

In civil aircraft it is usual practice to design the wing of sufficient strength to withstand the factored combined effects of 1g and 50 fps upward gust applied to the aircraft flying at its design cruising speed V_c mph. This safety factor is here assumed to be 1.5. By the use of Eqs. (3) and (4), then, the static strength requirement may be written

$$\text{Ultimate stress} = 1.5 (1 + \Delta n) \times 1g$$

$$= 1.5 \left(1 + \frac{F' a S \rho}{2W} \times 50 V_c \times 22/15\right) \times 1g \tag{5}$$

$$= n \times 1g$$

Here, n is defined as the ultimate acceleration factor or the ultimate load factor.

Eqs. (3) and (5) may be combined and simplified to yield the relation

$$v_h = \frac{75 \Delta n}{(n - 1.5)(V_{eh}/V_c)} \tag{6}$$

If a gust velocity v_h produces a stress $f\%$ of the ultimate, it readily follows that

$$f = \frac{\Delta n}{n} \times 100 \tag{7}$$

Substitution for Δn from Eq. (7) into Eq. (6) yields

$$v_h = Kf$$

$$= \frac{.75\, n}{(n - 1.5)(V_{eh}/V_c)} f \tag{8}$$

This expression establishes the relation between the gust velocity, the corresponding alternating stress, the ratio of the equivalent airspeed to the design cruise speed, and the ultimate load factor. The latter two terms are grouped together and defined as K in Eq. (8). It is apparent that if K is maintained constant, then the expression is a simple linear relationship between the percentage alternating stress f and the gust intensity v_h.

STRESS CYCLES FROM GUST DATA

The next step is to find an empirical relation between gust velocities of varying intensities being equalled or exceeded and the frequency with which these occur. Gust data are usually shown in two basic types of plots[5]. One shows the plot of miles flown before meeting a 10 fps gust or greater at varying altitudes. The other shows the plot of the number of gusts exceeding different magnitudes per one exceeding 10 fps. These data may be combined to obtain plots of miles m per gust against gust velocity v_h for different parametric values of the altitude. On a semi-log paper, these plots are straight lines in good approximation. The relation between m and v_h is therefore taken in the form

$$\log_{10} m = A v_h + B \tag{9}$$

as used by Chilver[1]. The constants A and B are determined from the given gust data. One such curve is shown in Figure 1 and is based on the gust data given by Raithby[5].

It follows from the definition of m (Eq. (9)) that the gust velocity v_h is equalled or exceeded $\frac{1}{m}$ times per mile of flight. Thus, Eq. (9) may be written in the form

$$\frac{1}{m} = \exp[-(A v_h + B) \log_e 10]$$

$$= \exp[-2.3 (A v_h + B)] \tag{10}$$

By the use of Eq. (8), the gust v_h in Eq. (10) may be replaced by the equivalent superimposed stresses of $f\%$ to yield

$$\frac{1}{m} = \exp[-2.3 (KAf + B)] \tag{11}$$

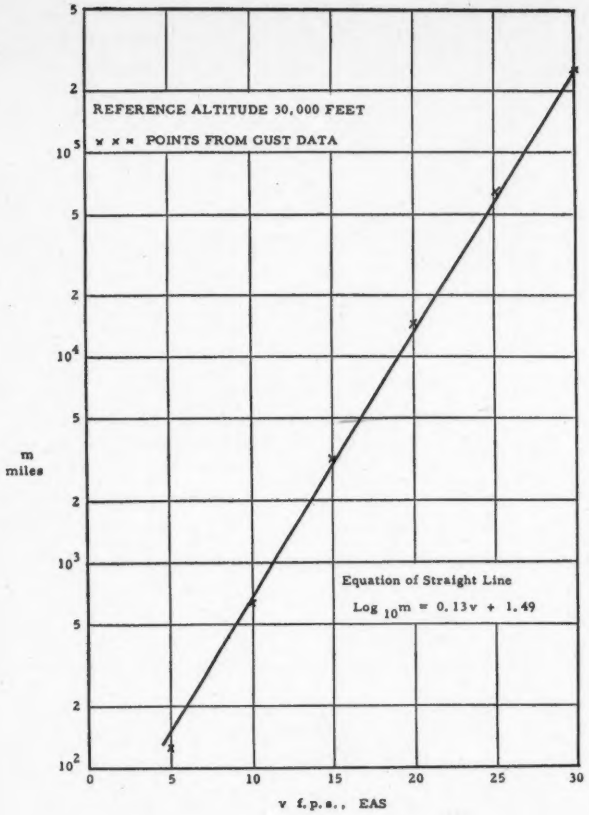


Figure 1

Gust velocity vs miles per gust over land

Since it has been assumed that there are an equal number of up and down gusts, the alternating percentage stress f is equalled or exceeded only $\frac{1}{2m} = E$ times per mile of flight. Thus

$$E(f) = \frac{1}{2} \exp [- 2.3 (KAf + B)] \qquad (12)$$

When the aircraft is flying at a velocity V_{eb}, the above equation defines the frequency with which the complete loading cycle $(1 \pm \Delta n) \times 1g = (1 \pm nf/100) \times 1g$ is equalled or exceeded.

AVERAGE S-N CURVE

Having obtained an expression for the frequency of the alternating stress cycles of varying intensity, the next step is to find a relation between alternating stresses σ_a and their number of applications $N(f)$ to failure. In order to establish this relation, it would be necessary to experimentally find the cycles to failure of a number of joints at various alternating stress levels. In general, there would be as many relations between σ_a and $N(f)$ (S-N curves) as there are joints. The use of a number of such relations would, naturally, lead to prohibitive mathematical complexities. Therefore, recourse is usually taken to the use of an average S-N curve which is based on a considerable number of typical aircraft joints test data.

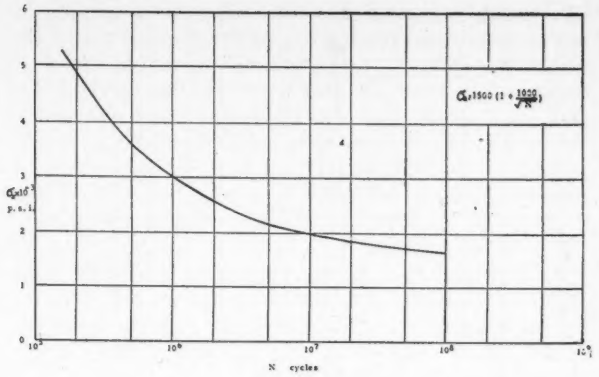


Figure 2

Average joint S-N curve

For such a curve Heywood[6] has suggested an approximate expression which may be written

$$\sigma_a = c + \frac{d}{\sqrt{N}} \qquad (13)$$

Here, c and d are experimental constants.

A typical curve of σ_a versus N is shown in Figure 2 for $c = 1.5 \times 10^3$ and $d = 10^6$ (the curve is reproduced from Reference 6).

Eq. (13) may be written in terms of f provided the ultimate stress σ_u is known. Thus, algebraically (see Eq. (7))

$$f = \frac{\Delta n}{n} \times 100 = \frac{\sigma_a}{\sigma_u} \times 100$$

$$= a + \frac{b}{\sqrt{N(f)}} \qquad (14)$$

where

$$a = 100c/\sigma_u = 1.5 \times 10^5/\sigma_u$$

$$b = 100d/\sigma_u = 1.5 \times 10^8/\sigma_u$$

Eq. (14) may be transposed to read

$$N(f) = \frac{b^2}{(f - a)^2} \qquad (15)$$

The frequency expression given by Eq. (12) and the cycles to failure expression given by Eq. (15) may now be used to analyze the fatigue life of an aircraft[a].

CUMULATIVE DAMAGE PER MILE

Although various theories exist for the purpose of evaluating the fatigue life of an aircraft, the cumulative damage theory has found wide use and appears to be more reasonable. The theory is comparatively simple and is based on the fundamental relation (Eq. (12)) between the number of cycles of each gust intensity $E(f)$ per mile and the corresponding alternating stress caused by the gust at a particular joint. If this alternating stress

[a]Eq. (13) suggested by Heywood[6] is for typical aircraft joints whose average mean stress is 12,800 psi. Should the mean stress of the aircraft considered differ considerably from this value, Eq. (13) may still be used for the purposes of analysis provided the final result is multiplied by the inverse ratio of the mean stresses for the actual aircraft life. (See Reference 7).

level causes failure of the joint after $N(f)$ number of applications then the ratio of $E(f)$ to $N(f)$ is a measure of the damage per mile caused at the joint by the particular stress cycle. The cumulative damage theory states that the total damage to the joint per mile of flight is the sum of the above ratios, each of which being associated with alternating stresses of different magnitudes (corresponding to varying gust intensities). The theory is discussed in incremental terms in the succeeding paragraphs.

The frequency $E(f)$ with which the complete stress cycle is equalled or exceeded is given by Eq. (12). It follows that for a small increment δf in f, the complete stress cycle is equalled or exceeded $E(f + \delta f)$ number of times per mile of flight. Thus

$$E(f) - E(f + \delta f) \tag{16}$$

represents the number of cycles applied to the structure in the range of stress cycles,

$$[1 \pm nf/100] \times 1g \text{ to } [1 \pm n(f + \delta f)/100] \times 1g \tag{17}$$

In this range there will be equal number of stress cycles having intensities greater than and less than $1 \pm n(f + \overline{\delta f})/100$ provided

$$E(f + \overline{\delta f}) = \frac{1}{2}[E(f) + E(f + \delta f)],\ \delta f > \overline{\delta f} > 0 \tag{18}$$

When an alternating stress cycle of intensity $1 \pm n(f + \overline{\delta f})/100$ is applied, the number of applications to failure is given by

$$N(f + \overline{\delta f}) \tag{19}$$

Since the wing structure is subjected to stress cycles of varying intensity under flying conditions, the cumulative damage theory assumes that the damage due to any alternating stress cycle is the ratio of the number of applied cycles of this stress to the number of cycles necessary for failure at the same stress. Hence, in the range f to $f + \delta f$ Expression (16) may be divided by Expression (19) to yield the infinitesimal damage

$$\delta\Delta = \frac{E(f) - E(f + \delta f)}{N(f + \overline{\delta f})} \tag{20}$$

so that

$$\frac{\delta\Delta}{\delta f} = -\frac{1}{N(f + \overline{\delta f})} \frac{E(f + \delta f) - E(f)}{\delta f} \tag{21}$$

Thus, in the limit as $\delta f \to \overline{\delta f} \to 0$, the damage rate per mile of flight is given by the simple expression

$$\frac{d\Delta}{df} = -\frac{1}{N(f)} \frac{d}{df} E(f) \tag{22}$$

If $N(f)$ and $E(f)$ are known as explicit functions of f, Eq. (22) may be integrated to obtain the total damage to the structure per mile of flight, due to any system of stresses alternating about the 1g stress.

Substituting into Eq. (22) the expressions for $N(f)$ and $E(f)$ given by Eqs. (12) and (15) yields

$$\frac{d\Delta}{df} = \frac{1}{2b^2}(f - a)^2 \exp[-2.3(KAf + B)] \tag{23}$$

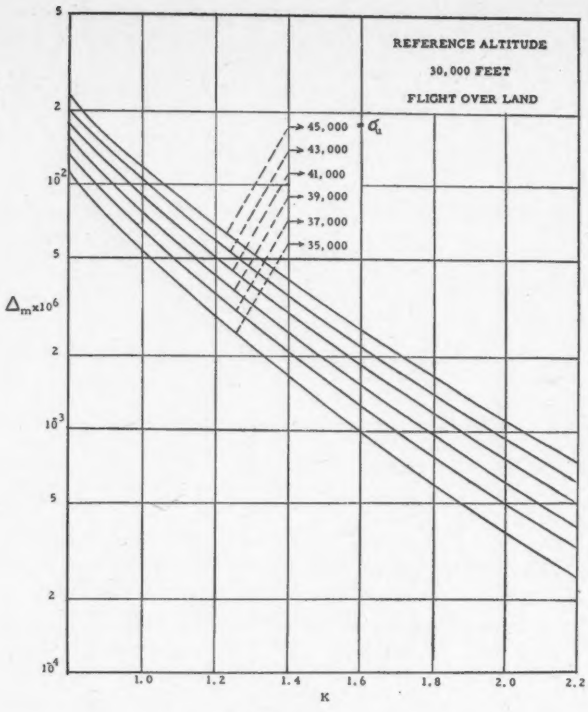


Figure 3

***K* vs damage per mile**

Examination of Eq. (15) shows that the minimum value of $f = a$ corresponds to $N \to \infty$. Eq. (23) may therefore be transformed to yield the particular integral

$$\Delta_m = \frac{1}{2b^2} \int_a^\infty (f - a)^2 \exp[-2.3(KAf + B)]\, df \tag{24}$$

This equation may be integrated by parts to yield the damage per mile

$$\Delta_m = \frac{e^{-2.3(KAa + B)}}{(2.3KAb)^2} \tag{25}$$

In the above equation the constants A and B are known, being determined from the given gust data. Since the experimental constant c and d are known (Eq. (13)), the values of a and b are readily computed from Eq. (14) for given values of the ultimate stress σ_u. Therefore, parametric curves of K versus Δ_m may be plotted for different values of the ultimate stress σ_u. Such plots are shown in Figure 3, being based on values of A and B corresponding to a reference altitude of 30,000 ft and a and b corresponding to $c = 1.5 \times 10^3$ and $d = 1.5 \times 10^6$ as given by Heywood[6]. The range of values for σ_u from 35,000 psi to 45,000 psi shown in the figure is usual in aircraft practice[b].

[b] A solution for the damage per mile was obtained by Eggwertz[4] in the form of a gamma-function where the S-N data were represented by a more general expression. However, for average aircraft joints, Heywood's[6] expression (Eq. (13)) is used in this paper as a sufficiently accurate representation of the S-N data.

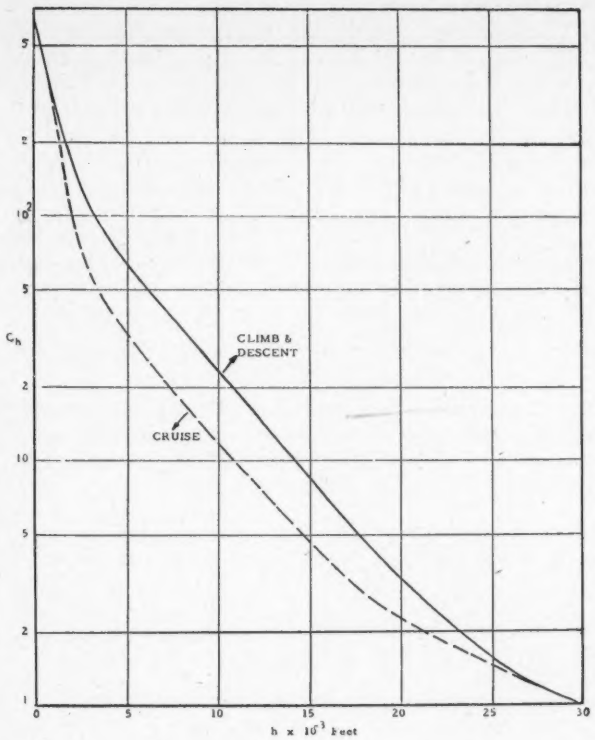


Figure 4
Altitude corrections

ECONOMICAL LIFE EVALUATION

Having established an expression for the damage per mile at a reference altitude, the economical life of an aircraft may be evaluated if the altitude climbed or descended, the horizontal speed of flight, and the horizontal distances travelled in the various stages of flight of an average mission are given.

For an aircraft flying at an altitude of h ft, the damage Δ_h in the infinitesimal time dt is readily seen to be

$$\Delta_h = C_h \Delta_{m_h} V_{t_h} \, dt \tag{26}$$

Here, the symbols C_h, Δ_{m_h}, and V_{t_h} represent, respectively, the altitude correction factor, the damage per mile, and the true airspeed (mph) at altitude h. From the gust data given by Raithby[5], the factor C_h obtained on a datum of unity at 30,000 ft is shown plotted against altitude in Figure 4.

The total damage in each stage of flight Δ_s may be obtained by integrating the right hand side of Eq. (26). Thus

$$\Delta_s = \int_{t_s} C_h \Delta_{m_h} V_{t_h} \, dt \tag{27}$$

where t_s represents the time required for the particular stage of flight. Now, C_h is a function of the altitude h and, in general, Δ_{m_h} and V_{t_h} are also functions of h for given equivalent airspeeds V_{e_h}. Therefore, in order to perform the integration in Eq. (27), it is necessary to transform the independent time variable t to the altitude variable h. To this end, it is convenient to analyze the flight mission in its different stages.

Damage during climb and descent

In the flight of a civil aircraft it is usual practice to climb and descend at a constant equivalent airspeed V_{e_c} and V_{e_d}, respectively. Subject to this assumption and a required static strength, Eq. (8) shows that K would have constant values through climb and descent. It follows from Eq. (25) that for a given gust data the damage per mile Δ_m is also constant, over climb and descent. Hence, by the use of Eq. (27), the damage during climb may be written

$$\Delta_c = \Delta_{m_c} V_{e_c} \int_{t_c} (C_h/\sigma_h^{1/2}) \, dt \tag{28}$$

where $(V_{e_c}/\sigma_h^{1/2})$ has been substituted for V_{t_h}. The term σ_h represents the ratio of the density of air at different altitudes to that at sea level. If the subscript c were replaced by d in Eq. (28), it would represent the damage during descent.

The transformation of the variable to h is carried out as follows. Up to 30,000 ft, usual in aircraft flight, the relative density h is given approximately by the relation

$$\sigma_h^{1/2} = 1 - 1.333 \times 10^{-5} \, h \tag{29}$$

Therefore, the true airspeed V_{t_h} may be written

$$V_{t_h} = V_{e_c}/\sigma_h^{1/2} = V_{e_c}/(1 - 1.333 \times 10^{-5} h) \tag{30}$$

The aircraft then travels a horizontal infinitesimal distance dd miles in the infinitesimal time dt, so that

$$dt = dd/V_{t_h} = dd(1 - 1.333 \times 10^{-5} h)/V_{e_c} \tag{31}$$

For given values of the ceiling climb altitude H_c ft and the corresponding horizontal distance D_c, the constant slope (Figure 5)

$$dd/dh = (D_c/H_c) \tag{32}$$

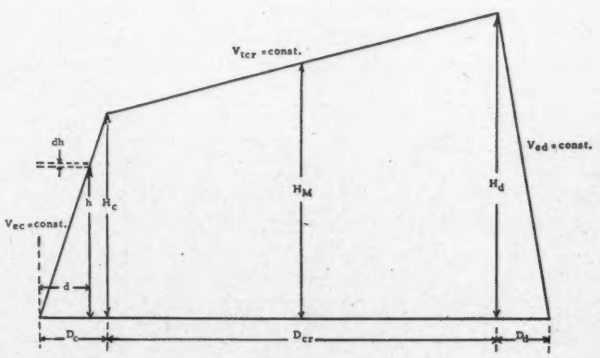


Figure 5
Average flight plan

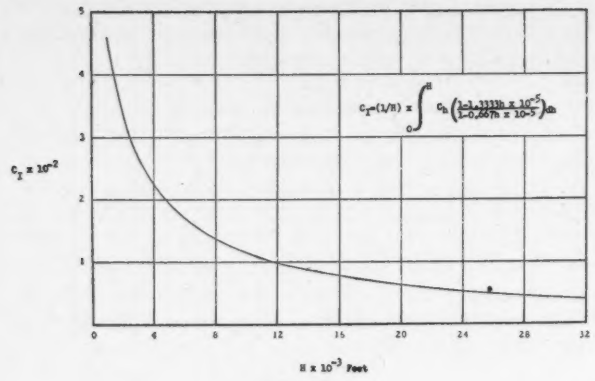


Figure 6

Integral altitude corrections

Thus, Eqs. (31) and (32) may be combined to obtain the incremental time-altitude relation

$$dt = \frac{(D_c/H_c)}{V_{ec}} (1 - 1.333 \times 10^{-5}h)\, dh \tag{33}$$

Integration of this equation shows the integration constant to be zero, since $h = 0$ at $t = 0$. Therefore,

$$t = \frac{(D_c/H_c)h}{V_{ec}} (1 - 0.667 \times 10^{-5}h) \tag{34}$$

Substitution of Eqs. (29) and (33) into Eq. (28) and simplification yield the damage during climb in a flight mission

$$\Delta_c = \Delta_{mc} \times D_c \times (1/H_c) \int_0^{Hc} C_h \left(\frac{1 - 1.333 \times 10^{-5}h}{1 - 0.667 \times 10^{-5}h}\right) dh \tag{35}$$

Similar analysis yields for the damage during descent

$$\Delta_d = \Delta_{md} \times D_d \times (1/H_d) \int_0^{Hd} C_h \left(\frac{1 - 1.333 \times 10^{-5}h}{1 - 0.667 \times 10^{-5}h}\right) dh \tag{36}$$

Examination of Eqs. (35) and (36) shows that the integrand is only a function of the atmospheric conditions, while the first two terms apply to a particular aircraft on an average flight mission. It is difficult to evaluate the particular integrals analytically, since the altitude correction factor C_h is not known as an explicit function of the altitude h. However, the integrals may be readily evaluated by a numerical procedure by the use of the C_h versus h curve (Figure 4). The integrals were evaluated by the use of Simpson's Rule for various values of the upper limit H and Figure 6 shows a plot of

$$H \text{ versus } (1/H) \int_0^H C_h \left(\frac{1 - 1.333 \times 10^{-5}h}{1 - 0.667 \times 10^{-5}h}\right) dh$$

It is apparent that the curve could be used for both climb and descent.

Damage during cruise

In contrast to climb and descent, aircraft are generally maintained at constant true airspeeds V_{tcr} during cruise. The evaluation of the damage to an aircraft during this stage, naturally, differs from that discussed for climb and descent. The equivalent airspeed V_{eh} would now vary with the altitude h, the relation being given by (see Eq. (30))

$$V_{eh} = V_{tcr}(1 - 1.333 \times 10^{-5}h),\ H_c < h < H_d \tag{37}$$

Therefore, Eq. (8) shows that K would also vary with the altitude and may be expressed as

$$K = \frac{0.75nV_c}{(n - 1.5)\, V_{tcr}} \times \frac{1}{(1 - 1.333 \times 10^{-5}h)},\ H_c < h < H_d \tag{38}$$

It follows from Eq. (25) that the damage per mile during cruise would vary with the altitude. Hence, by the use of Eq. (27), the damage during cruise may be written

$$\Delta_{cr} = V_{tcr} \int_{t_{cr}} C_h\, \Delta_{mh}\, dt \tag{39}$$

Based upon the assumptions of a constant true airspeed V_{tcr} during cruise, Eqs. (31) and (32) may be combined to yield the linear relation

$$t = \frac{D_{cr}\, h}{V_{tcr}(H_d - H_c)} \tag{40}$$

between time and altitude, where D_{cr} is the horizontal distance in miles travelled during cruise. Hence, the differential relation for the cruise stage is

$$dt = \frac{D_{cr}}{V_{tcr}(H_d - H_c)}\, dh \tag{41}$$

Substitution of Eq. (41) in Eq. (39) and simplification yield the damage during cruise

$$\Delta_{cr} = \frac{D_{cr}}{(H_d - H_c)} \int_{H_c}^{H_d} C_h \Delta_{mh}\, dh \tag{42}$$

where Δ_{mh} may be written as an explicit function of h by the use of Eq. (8) and Eq. (25). The integral may then be evaluated again by Simpson's Rule.

It seems appropriate to mention here that for aircraft cruising at relatively high altitudes (about 20,000 ft to 30,000 ft or above), the numerical evaluation of the somewhat complex integral may be dispensed with and in good approximation an alternative procedure used to compute the damage during cruise. In such a procedure, the equivalent airspeed V_{eh} (Eq. (37)) and the corresponding K (Eq. (38)) are evaluated for the mid-altitude $h = H_M = (H_d + H_c)/2$ of cruise. The damage per mile Δ_{H_M} is then readily computed for this one value of K. It is then seen from Eq. (42) that the damage to the aircraft during cruise in a flight mission is

$$\Delta_{cr} = V_{tcr}\, C_{H_M}\, \Delta_{mH_M} \int_{t_{cr}} dt$$

$$= V_{tcr}\, C_{H_M}\, \Delta_{mH_M}\, t_{cr} \tag{43}$$

where the values C_{H_M} and $\Delta_{m_{H_M}}$ correspond to the mid-altitude of cruise and t_{cr} is total cruise time. Substitution for t_{cr} from Eq. (40) yields

$$\Delta_{cr} = C_{H_M}\, \Delta_{m_{H_M}}\, D_{cr} \tag{44}$$

This value of the damage is approximately the same as that given by Eq. (42) for relatively high cruise altitudes (about 20,000 ft to 30,000 ft).

Economical life

With the damages in each stage evaluated by the use of Eqs. (35), (36) and (42) or (44), they are summed to give the total damage during one flight mission

$$\Delta_{tot} = \Delta_c + \Delta_{cr} + \Delta_d \tag{45}$$

The failure of the aircraft occurs when Δ_{tot} times the number of aircraft flight missions F_n reaches unity. Hence, theoretically, the number of flight missions for failure of the aircraft is given by

$$F_n = 1/\Delta_{tot} \tag{46}$$

If t_{tot} is the total time of each flight mission, it follows that the theoretical value of the economical life is given by

$$\begin{aligned} L_e &= F_n\, t_{tot} \\ &= F_n\, (t_c + t_{cr} + t_d) \end{aligned} \tag{47}$$

Eqs. (34) and (40) show that

$$\left.\begin{aligned} t_c &= (D_c/V_{e_c})\,(1 - 0.667 \times 10^{-5}\, H_c) \\ t_{cr} &= (D_{cr}/V_{t_{cr}}) \\ \text{and} \\ t_d &= (D_d/V_{e_d})\,(1 - 0.667 \times 10^{-5} H_d) \end{aligned}\right\} \tag{48}$$

In order to obtain a more representative value of the actual economical life, account must be taken of the scatter in the test data and the uncertainty of the gust variations. Furthermore, some allowance must be made for the difference between the mean fatigue life N_m of the test specimen and the cycles to failure N_f of the average joint.

The factor f_s to allow for the scatter in the test data would, of course, depend upon the number of specimens tested and would reduce the theoretical value of the economical life. Similarly, the factor f_g to allow for the uncertainty of the gust variations further reduces the life of the aircraft. The factor to allow for the difference in the mean test life N_m and the average joint life N_f at the same stress level is given by the ratio N_m/N_f. These practical considerations may be summarized in the form of an overall correction factor

$$f_s = \frac{(N_m/N_f)}{f_s f_g} \tag{49}$$

Finally, the corrected economical life of the aircraft may be written (see Eqs. (46) and (47))

$$L_{e_c} = f_c\, L_e = f_c\, t_{tot}/\Delta_{tot} \tag{50}$$

Example

The economical life analysis of an aircraft discussed in the preceding sections is now illustrated by a typical example. The average flight plan, shown schematically in Figure 5, is assumed to be represented by the following given values:

(1) Climb to an altitude $H_c = 20{,}000$ ft at a constant equivalent airspeed $V_{e_c} = 200$ mph over a distance $D_c = 100$ miles.

(2) Cruise from an altitude $H_c = 20{,}000$ ft to an altitude $H_d = 30{,}000$ ft at a constant true airspeed $V_{t_{cr}} = 360$ mph over a distance $D_{cr} = 3{,}600$ miles.

(3) Descend from an altitude $H_d = 30{,}000$ to zero ft at a constant equivalent airspeed $V_{e_d} = 280$ mph over a distance $D_d = 80$ miles.

The following additional information is assumed to be given:

Design maximum sea level flight speed $V_e = 300$ mph
Ultimate load factor $n = 4.15$
Ultimate stress (static strength) $\sigma_u = 40{,}000$ psi
Mean life of specimen tested $N_m = 0.8 \times 10^6$
Test scatter factor $f_s = 4.5$
Gust scatter factor $f_g = 1.5$

By the use of Eq. (8), the value of K during climb may be computed as

$$K_c = \frac{0.75 \times 4.15}{(4.15 - 1.5)\,(200/300)}$$

$$= 1.76$$

For this value of K, the damage per mile on the $\sigma_u = 40{,}000$ psi curve in Figure 3 reads as

$$\Delta_{m_c} = 0.00116 \times 10^{-6}$$

The integral altitude correction for a climb of $H_c = 20{,}000$ ft is read from Figure 6 as

$$C_{I_c} = 63$$

Since the horizontal distance travelled during climb is $D_c = 100$ miles, the damage during climb per flight is given by Eq. (35) as

$$\Delta_c = 0.00116 \times 10^{-6} \times 100 \times 63$$

$$= 7.30 \times 10^{-6}$$

Similarly the damage during descent per flight is

$$\Delta_d = 0.00400 \times 10^{-6} \times 80 \times 43$$

$$= 13.74 \times 10^{-6}$$

Now Eq. (42) is used to compute the damage during cruise. The particular integral in the equation is evaluated by Simpson's Rule to yield the result 32.68. Hence, the equation gives the damage during cruise per flight as

$$\Delta_{cr} = \frac{3600 \times 32.68}{10000}$$

$$= 11.75 \times 10^{-6}$$

In order to emphasize the usefulness of the approximate formula (Eq. (44)), the damage during cruise is also computed on the basis of this approximation. For the average altitude during cruise $H_M = 25{,}000$ ft, the value K_{cr} is readily computed to be 1.47 and from Figure 3 the corresponding $\Delta_{m_{cr}} = 0.00229 \times 10^{-6}$. The altitude correction factor $C_{H_M} = 1.45$ is read from Figure 4. Hence, Eq. (44) gives the damage during cruise per flight as

$$\Delta_{cr} = 0.00229 \times 10^{-6} \times 3600 \times 1.45$$

$$= 11.97 \times 10^{-6}$$

This value differs from the exact value by about 2%.

Finally, the total damage per flight is given by

$$\Delta_{tot} = \Delta_c + \Delta_{cr} + \Delta_d$$
$$= 32.79 \times 10^{-6}$$

so that the number of flights to failure is

$$F_n = \frac{10^6}{32.79}$$
$$= 30{,}500$$

The elapsed time during climb, cruise and descent is computed by the use of Eq. (48) to be

$$t_{tot} = 10.538 \text{ hours}$$

From Eq. (15), $N_f = 10^6$ for a standard value of $f = 7.5$. Since N_m, f_s and f_g are given, respectively, as 0.8×10^6, 4.5, and 1.5, the actual economical life of the aircraft is

$$L_{ec} = \frac{30{,}500 \times 10.538 \times 0.8}{4.5 \times 1.5}$$
$$= 38{,}200 \text{ hours.}$$

CONCLUSION

A relatively simple method of aircraft life evaluation has been presented. The method is quick and may prove useful in obtaining sufficiently accurate estimates of the economical life of an aircraft. Before the validity of the method is accepted, a few brief remarks seem in order.

The different variables involved in aircraft flight complicate its fatigue life analysis. For instance, the various joints in an aircraft may be operating at different stress levels. Even a complex analysis to account for each of these can only provide an approximate estimate of its fatigue life. Further, intermittent changes in flight plans for practical reasons would, obviously, alter any estimate of life. However, comparisons of aircraft life are necessary, and it is therefore hoped that the simple method presented in this paper proves useful in obtaining aircraft economical life in good approximation.

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TURBOJET LIFT ENGINE DESIGN FOR VTOL TRANSPORTS†

by M. J. Saari*

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SUMMARY

The pure hoverjet, utilizing separate turbojet engines for lift and cruise phases of flight, is an interesting high speed VTOL transport airplane configuration. The results of a brief analysis are presented with the intent to: (1) define the effects of some general powerplant installation and operating variables on VTOL transport performance; (2) indicate desirable turbojet lift engine design characteristics with respect to compressor pressure ratio, turbine inlet temperature, component efficiencies and thrust augmentation; and (3) examine the unique lift engine operating requirements for feasible ways of reducing engine specific weight. The figure of merit used to gauge airplane performance is airplane range.

INTRODUCTION

THE salient virtues of VTOL transport aircraft have been discussed in many publications and presentations[1, 2, 3, 4, 5]. A cursory review of some of these assigned virtues relative to conventional type aircraft indicates that the VTOL transport promises to (1) permit operation from limited or unprepared areas, (2) reduce, if not eliminate, holding requirements during adverse weather or traffic conditions at air terminals, (3) provide added margin of safety in flight emergency situations, (4) reduce noise and hazards associated with low level approaches over populated areas adjacent to airports, and (5) allow greater freedom and economy in the selection and construction of air terminal sites.

It would indeed be naive to assume that the VTOL transport has clear-cut advantages, such as these, in comparison with the conventional transport, for experience teaches us that a price is always attached to certain luxuries. Thus, instead of thinking in a competitive sense, it seems better to consider the VTOL transport as a different mode of transportation that performs a specialty function of augmenting existing and future transportation systems.

Design concepts for transport aircraft possessing vertical takeoff and landing capabilities have been numerous and diverse[6]. In the interests of developing a perspective for this discussion, a brief recapitulation of the general categories of proposed VTOL configurations is in order. The rotating wing aircraft utilizing a reciprocating or gas turbine type powerplant is represented by the helicopter. Aircraft employing gas turbine driven propellers that provide both hover and cruise propulsion include the tail sitter, tilt-wing or tilt-engine, the shrouded propeller, deflected slip stream and wingless aerodyne configurations. Examples of design concepts incorporating separate powerplants for hovering and cruise flight are the compound helicopter, retractoplane, and ducted propeller-turbojet combination aircraft. The final category consists of the pure hoverjets that derive hover and cruise power from turbojet engines in either tilt-wing or tilt-engine configurations where lift and cruise power are supplied by the same engines or in configurations having separate fixed attitude lift and cruise engine installations.

In a broad sense, there is a configuration that performs a specific VTOL mission better than any other configuration since each geometry has definite performance limitations with regard to hovering capability, range, payload capacity or maximum cruising speed. The VTOL transport configuration selected for this analysis was the high subsonic hoverjet utilizing separate engines for hover and cruise flight. This selection was based on the assumption that an interest exists or at least will be developed in commercial VTOL transports. Thus, in order to be compatible with the trend in modern air transportation, the VTOL transport must have a high cruise speed. Specification of cruise Mach numbers greater than about 0.60 precludes practical consideration of those VTOL configurations dependent on propellers for cruise propulsion due to limitations of propeller aerodynamics. On the other end of the speed scale, it has been shown by other analyses[3] that the turbojet-powered supersonic VTOL transport must be quite large

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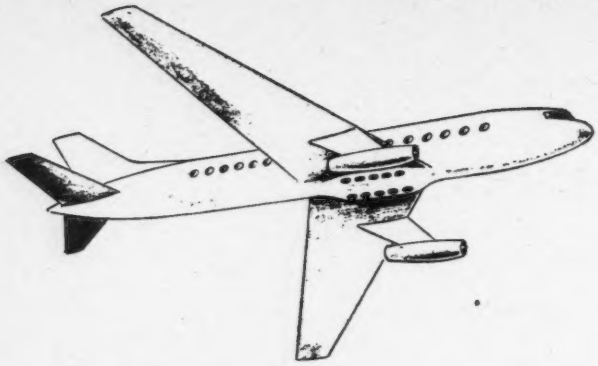

GROSS WEIGHT, 50,000 LB
FUSELAGE FINENESS RATIO, 10
WING LOADING, 150 LB/SQ FT
WING ASPECT RATIO, 6
WING SWEEP, 35°
CRUISE LIFT-DRAG RATIO, 15
CRUISE SPEED, MACH 0.9 AT 30,000 FT

Figure 1
Assumed VTOL jet transport configuration

and must cruise at very high altitudes to obtain the lift/drag ratios required for acceptable range/payload performance. This type of aircraft has been suggested for long range operation where block time savings are significant.

The intermediate choice narrows down to a subsonic VTOL transport having a cruise Mach number of about 0.90. The inherently high lift/drag ratios associated with subsonic speeds imply the feasibility of obtaining reasonable range/payload performance with moderate airplane gross weights. The specification of high cruise speed infers the necessity of using turbojet engines for cruise propulsion.

Since VTOL capability requires installed thrust levels greater than airplane gross weight, it is apparent that, all other factors being equal, the lifting powerplant having the least weight per unit of thrust produced (lowest specific weight) would provide the best overall airplane performance. Lifting powerplant systems such as ducted propellers buried in the wings and driven by tip turbines or geared turbines provide good thrust to power ratios depending upon the disk loading employed, but little is known at the present time about the actual thrust to weight ratios that can be achieved. Advancements in turbojet engine design, on the other hand, have produced engines with specific weights approaching 0.10. Even lower specific weights may be realized by careful consideration of specific lift engine operating requirements as will be discussed later. The lightweight turbojet engine was therefore selected as the lifting powerplant. The selection of the pure hoverjet for this analysis should not be construed as an exclusion of other geometries for transport configurations.

The specific intention of this analysis was to (1) determine the effects of some of the general powerplant installation variables on VTOL transport performance, (2) indicate desirable lift engine design characteristics for VTOL transport applications, and (3) examine the unique operating requirements of the lifting turbojet engines for practical approaches toward reducing engine specific weight.

The performance of the VTOL jet transport is compared with that of an equivalent conventional transport to provide a reference evaluation.

ASSUMED AIRCRAFT SPECIFICATIONS

The VTOL jet transport configuration assumed for this analysis is illustrated in Figure 1. Some of the basic characteristics included a fuselage fineness ratio of 10.0, wing aspect ratio of 6.0, wing sweep of 35° and a wing loading of 150 lb/sq ft. The high wing loading is justified by the fact that takeoff and landing are not dependent on wing lift so that the wing area can be sized to favour the cruise condition. The overall airplane cruise L/D was computed to be 15.0. The flight plan was assumed to consist of (1) vertical takeoff, (2) transition to wing-supported flight, (3) climb to cruise altitude of 30,000 ft, (4) Breguet cruise at Mach number of 0.90, (5) let-down and transition to hover condition, and (6) vertical landing.

The equivalent conventional jet transport was assumed to have the same basic characteristics except that the wing loading was 100 lb/sq ft. The overall airplane lift/drag ratio is not greatly affected by the range of wing loading considered here so the conventional airplane was also assumed to have a cruise L/D of 15.0. The engines were sized to give a gross weight to thrust ratio of 4.0 and were assumed to have the same specific weight as the cruise engines of the VTOL transport. The assigned cruise altitude and flight Mach number were 30,000 ft and 0.90, respectively.

The assumed weight breakdowns for both airplanes are shown in Figure 2. The takeoff gross weight of each was 50,000 pounds (about the size of the Viscount transport). The payloads, structural weights, fixed weights and cruise engine weights were assumed to be equal. Although the wing and landing gear weights were lower for the VTOL airplane, the additional ducting, control systems and accessory equipment required in the VTOL airplane were considered to have a balancing effect. The fuel reserve allowance for the conventional airplane was 5% of the gross weight or about 13% of the total fuel weight for the reference condition. The fuel tank weight was about 2% of the gross weight in

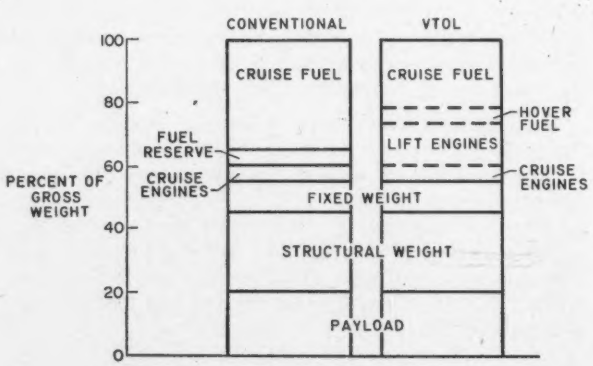

Figure 2
Typical assumed weight breakdown

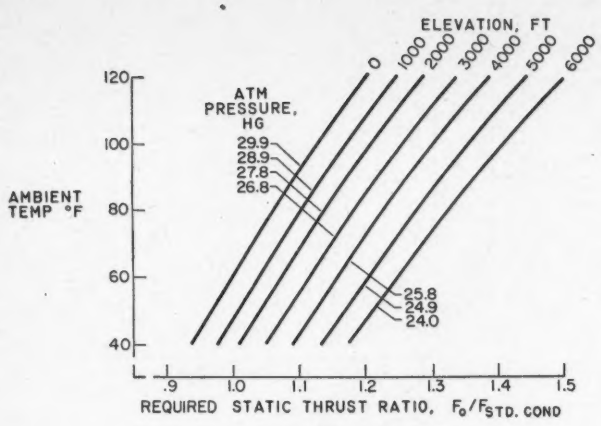

Figure 3
Effect of atmospheric conditions on required static thrust ratio

each case. The cruise fuel weight for the VTOL transport was considered a variable dependent upon installation and operating variables. The figure of merit used to express airplane performance throughout the discussion is airplane range.

AIRCRAFT PERFORMANCE

The performance of the hoverjet transport is, of course, a function of many variables, a full treatment of which is beyond the scope of this paper. However, by restricting the analysis to a specific airplane and flight plan, it is feasible to define the effects of some of the more influential variables related to the powerplant installations on airplane performance and, in particular, those variables dealing with the hovering and transition portions of the flight plan. The powerplant considerations, in essence, consist of two groups of variables; one group that deals with the requirements and consequences of vertical flight, and the other that defines the desirable powerplant design characteristics for the assumed application. The effects on range performance of some of the variables in each of these groups will be discussed each in turn for fixed values of the other related variables. Unless otherwise stated, the basic reference values of the variables involved in the analysis will be as follows:

(1) Atmospheric conditions – sea-level NACA standard-day
(2) Takeoff thrust to gross weight ratio – 1.25
(3) Total hovering time – 2 minutes
(4) Reference lift engine specific weight – 0.10 and 0.20
(5) Cruise engine specific weight – 1.30 × reference lift engine specific weight
(6) Compressor pressure ratio – 8.0:1
(7) Turbine inlet temperature – 2,000°R
(8) Compressor efficiency – 90%
(9) Combustion efficiency – 95%
(10) Turbine efficiency – 85%
(11) Thrust deflection efficiency – 90%

The cruise engine characteristics (items (6) to (10)) were held constant throughout the analysis. In addition, it was assumed that full cruise engine thrust was deflected 90° to contribute to the lifting force; afterburner weight (where considered) was assumed to be 15% of the bare-engine weight. The engine installation weight factors were considered to be 1.25 for the lift engines and 1.20 for the cruise engines. In cases where the compressor pressure ratio was variable, the engine specific weight was considered a function of pressure ratio.

Installation and operational variables

Ambient pressure and temperature

A basic consideration in defining the installed thrust required for vertical flight is the atmospheric environment in which the engine must operate. The effects of ambient pressure and temperature on static thrust ratio required to maintain design lifting thrust are shown in Figure 3. At standard-day conditions, the static thrust ratio is unity by assignment. An increase in either ambient temperature or elevation causes the thrust of a turbojet engine to decrease due to reduced airflow and engine pressure level. Thus, if a given thrust to gross weight ratio is to be maintained, the installed thrust must be increased. For example, when the sea-level temperature is 100°F the installed thrust required must be increased by 13% and if in addition the elevation were increased to 4,000 ft, the installed thrust must be 30% higher than the design sea-level thrust if design thrust to gross weight ratio is to be conserved. Therefore, in fixing the thrust to gross weight ratio in the airplane design, the tendency would be to select a sufficiently large value to insure adequate thrust margin under the most adverse operating conditions anticipated. There are, however, other considerations that influence this decision.

Thrust to gross weight ratio

The selection of a design thrust to gross weight ratio must be tempered by the effect of this variable on overall airplane performance as shown in Figure 4. The indications are that a 10% increase in thrust to gross weight ratio results in about an 8% reduction in range for the light engines and about a 28% reduction for the heavier engines. These range reductions are caused primarily by the increase in total lift engine weight and partly by the increase in fuel consumed during the

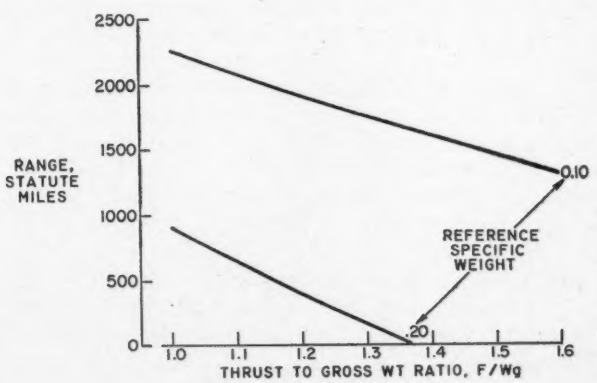

Figure 4
Effect of lifting thrust to gross weight ratio on range

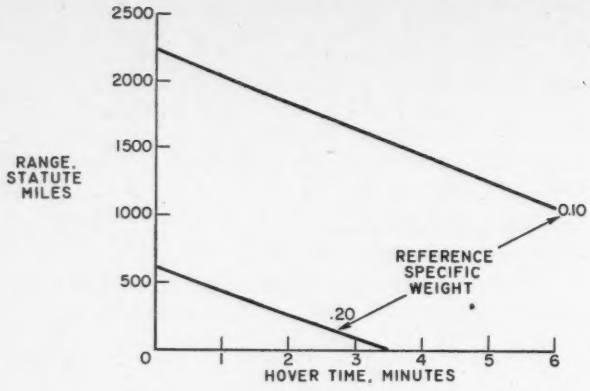

Figure 5
Effect of hover time on range

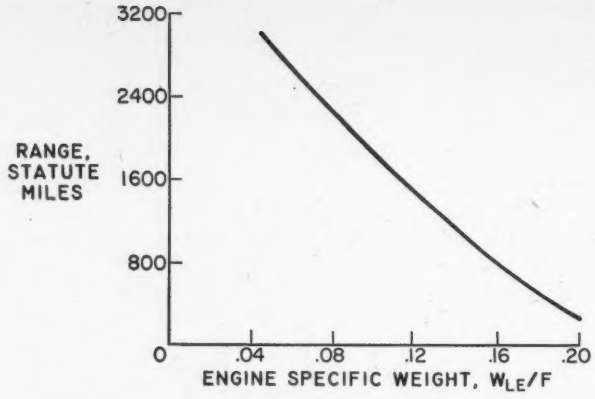

Figure 6
Effect of lift engine specific weight on range

hovering condition, both of which detract from the fuel weight available for cruise. It is important, therefore, that the design takeoff thrust to gross weight ratio be compromised between the value for maximum range and that dictated by the most severe atmospheric conditions anticipated in aircraft operation. Thrust augmentation by water injection may be used to alleviate this problem if the weight of the associated equipment required, such as pumps, tankage, controls etc, is less than the weight added by extra lightweight engines. A thrust to gross weight ratio of 1.25 was selected for this analysis as being a reasonable value with the assumption that under extremely adverse conditions a sacrifice in payload or range is acceptable.

Hover time

The fact that the lifting thrust level for vertical flight is high means that the total rate of fuel consumption is correspondingly high. It would therefore be expected that the total time spent in the vertical and transition phases of flight would have an appreciable effect on airplane range. The relationship between hover time and range is shown in Figure 5. For the airplane considered herein, each additional minute of hovering time reduces the range capability by about 200 miles. Thus, it is important that the total hover time be kept to a minimum if range is to be conserved. A preliminary analysis indicated that the takeoff, transition and landing portions of the flight plan can be reasonably accomplished in a total time of two minutes. It is apparent that the hover time requirement is directly related to the design thrust to gross weight ratio in that a high value of the latter yields a short takeoff time. In the final analysis, the minimum hover time is governed by the vertical acceleration tolerance of the aircraft structure if not by the intestinal structure of the paying passengers.

The foregoing represent some of the gross lifting powerplant installation design considerations. The second group of variables – those dealing with specific turbojet engine design characteristics – are discussed in the following.

Engine design variables

As was mentioned earlier, VTOL capability requires installed thrust levels greater than airplane gross weight. Since the cruise fuel supply is interchangeable with engine weight, it is of interest to evaluate the effect of this trade on airplane performance. The criticality of lift engine specific weight with respect to airplane range performance is shown in Figure 6. The extreme slope of the curve indicates, first, that lift engine specific weight must be low in order to realize acceptable range performance and, secondly, that efforts toward reducing lift engine specific weight are to be rewarded by sizeable improvements in range. The fact that turbojet engines having specific weights approaching 0.10 have been operated successfully gives credence to the belief that VTOL jet transports having reasonable range/payload performance are feasible within the existing state of the art.

The cruise engine specific weight as shown in Figure 7 also affects airplane range, but in a more diminutive way since the cruise engine thrust and weight are relatively small in comparison with the aggregate lift engine thrust and weight. Though a less critical factor, cruise engine specific weight is nevertheless important in accumulative performance gains. Here, again, it is interesting to note that turbojet engines suitable for VTOL cruise applications, having specific weights of about 0.14, already exist as operating hardware.

The fact that engine specific weight has a great influence on airplant performance is interesting but not completely unexpected. The real question at hand is: "What characteristics must a turbojet engine embody to be singularly suited for VTOL jet transport applications?" Once these characteristics are defined, a discrete

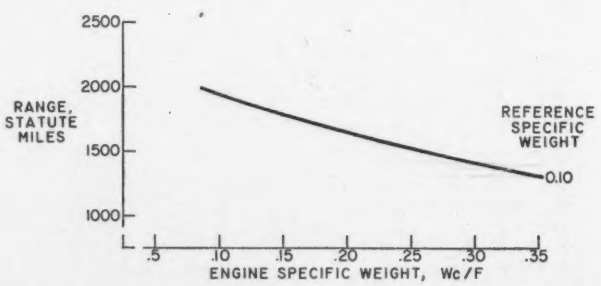

Figure 7
Effect of cruise engine specific weight on range

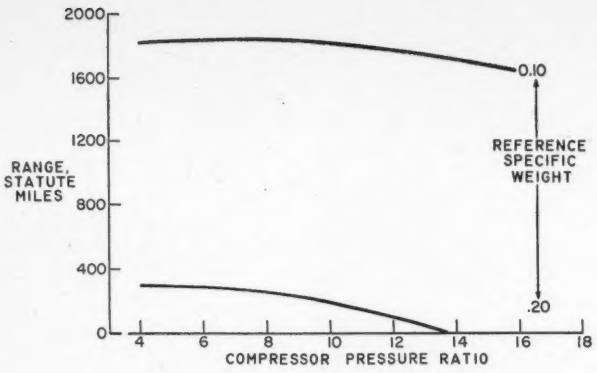

Figure 8
Effect of lift engine compressor pressure ratio on range

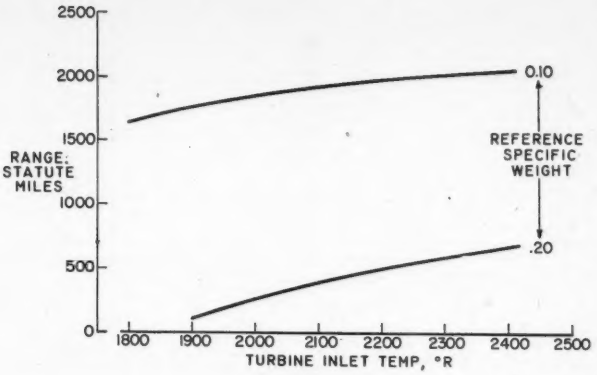

Figure 9
Effect of lift engine turbine inlet temperature on range

assessment may be possible that will suggest a composite engine design that approaches the optimum, which, in this case, is an engine having a minimum specific weight.

The succeeding discussion will be limited to the lift engine characteristics consisting of compressor pressure ratio, turbine inlet temperature, component efficiencies and augmentation by afterburning. The cruise engine characteristics need not be considered here since the design is, by and large, governed by the same well known considerations as for conventional subsonic jet aircraft.

Lift engine compressor pressure ratio

In turbojet engine design, it is generally accepted that overall compressor pressure ratio not only governs specific fuel consumption through cycle efficiency, but also affects engine weight through both the number of stages and the general engine structure required by the internal pressure levels to be attained. Thus, for high pressure ratios the specific fuel consumption is low but the engine weight tends toward the high side while at low pressure ratios the reverse is true. Any approach to a realistic analysis of the effect of compressor pressure ratio on airplane range, such as shown in Figure 8, should include an accounting for the variation of engine specific weight and specific fuel consumption with compressor pressure ratio. For the case considered here it was assumed that the specific engine weight corrections for compressor pressure ratios of 4, 8, 12 and 16, were 0.875, 1.0, 1.07 and 1.11 respectively and the specific fuel consumption corrections were 1.28, 1.0, 0.90 and 0.83 respectively. The data indicate that under the assumed conditions, airplane range is rather insensitive to lift engine compressor pressure ratio particularly for the lower reference specific weight. The selection of the lift engine design compressor pressure ratio therefore reverts to a consideration of economy and produceability so that the apparent choice would be a low pressure ratio machine. It should be noted that, since these data apply to conventional compressor stage loadings, it can be expected that some improvement in engine specific weight and hence airplane range may be achieved by the reduction of the number of compressor stages associated with increased stage loadings. This aspect will be considered later in the discussion.

Turbine inlet temperature

Analysis of the gas turbine cycle indicates that all other factors being constant, the thrust and hence the specific weight is improved by increasing turbine inlet temperature. The data in Figure 9, showing range as a function of turbine inlet temperature, leads to the conclusion that elevated temperatures improve range but the gains for reasonable temperature increments above accepted present-day values (about 2,000°F) are relatively small, particularly for the lighter engines. The apparent reason for the relative insensitivity of range to turbine temperature elevation is that the increase in specific fuel consumption nearly balances the improvement in effective engine specific weight gained by upgrading the thrust. Attempts to increase turbine inlet temperature must contend with adverse effects of elevated temperatures on stress rupture and thermal shock tolerances of the turbine disk and blading. Since time-failure characteristics are unique for each turbine design, the ultimate assignment of temperature level remains a revelation of specific engine development. In connection with turbine operating temperature, it is doubtful that any significant overall performance gains can be obtained by the added complication of turbine cooling.

Component efficiencies

The effects of lift engine compressor, combustor and turbine efficiencies on airplane range are presented in Figure 10. Variation of compressor or turbine efficiency

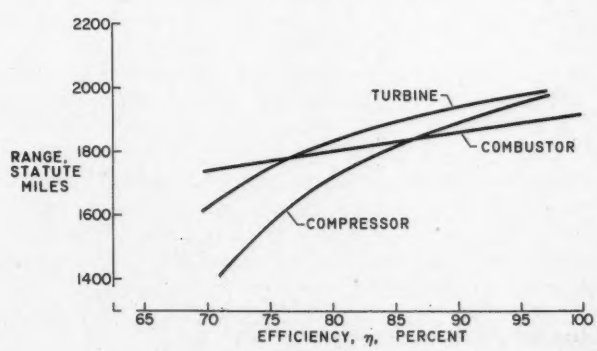

Figure 10
Effect of component efficiencies on range

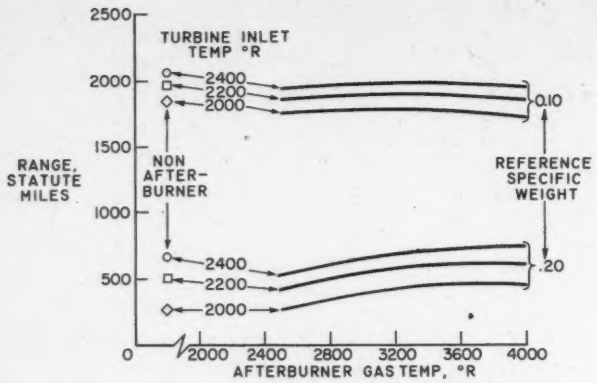

Figure 11
Effect of thrust augmentation by afterburning on range

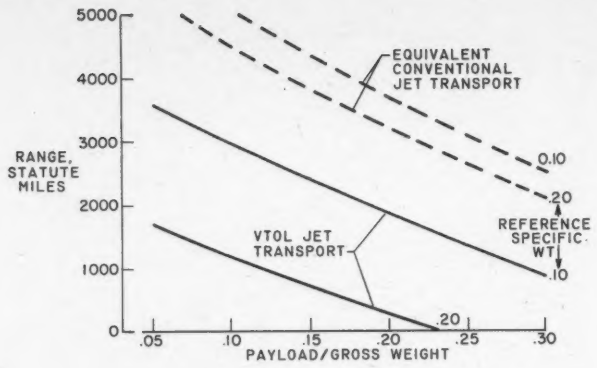

Figure 12
Comparison of performance of VTOL transport with equivalent conventional transport

affects both specific fuel consumption and thrust, the latter being associated with engine specific weight. The data indicate that a one point change in the efficiency of either the compressor or turbine component is reflected in roughly a 1% change in range within the region of efficiencies of practical interest. A similar variation in combustion efficiency, which incidentally affects only the specific fuel consumption, causes a 0.3% change in range.

While the effects of the individual component efficiencies on range might be considered small, a complacent attitude in this regard should be avoided, for the accumulative effect could be quite significant. Furthermore, design factors which affect component efficiencies also affect overall engine operating stability. These effects should rather be kept in mind in making gainful compromises in engine design in the interest of making possible reductions in engine weight or in the easement of production problems.

Thrust augmentation by afterburning

The augmentation of lift engine thrust by afterburning is outwardly a feasible method of reducing effective engine specific weight which, in effect, corresponds to a reduction in the number of engines required for vertical flight. The effect of afterburner gas temperature on range performance is shown in Figure 11 together with range obtained by dry engine operation for several turbine inlet temperatures. The general conclusion drawn from these data is that the afterburner does not improve range performance and, in fact, for the light-weight engines, imposes a performance penalty relative to the nonafterburning engine configuration. The added weight of the afterburner combined with a twofold increase in fuel consumption outbalances the effects of increased thrust. In addition, the high exhaust temperatures, noise level and the complications in the fuel system, control system and exhaust nozzle variation required all contribute to the undesirability of incorporating afterburners in the hoverjet transport.

Airplane performance comparison

Although the VTOL airplane represents a separate and unique entity in the field of air transportation and since earlier in the discussion it was mentioned that there is obviously a price tag attached to VTOL capability, human frailty prevails. The magnitude of this price in terms of payload and range is illustrated in Figure 12, where the performance of the assumed VTOL jet transport is compared with that of an equivalent conventional-type jet transport. For the reference engine specific weight of 0.10, the range penalty for VTOL capability is between 1,950 and 1,650 miles for payload to gross weight ratios between 0.10 and 0.30, respectively. This range decrement results primarily from the trade of lift engine weight with cruise fuel capacity. A comparison of the relative effects of engine weight on the range performance of both aircraft emphasizes again the tremendous incentive to search out ways and means of reducing the lifting powerplant specific weight.

LIFT ENGINE DESIGN

The foregoing discussion has dealt with some general as well as specific powerplant consideration relative to the VTOL jet transport. While many subtleties remain unheralded, at least the gross design requirements were illuminated. The more ambiguous part of lift engine design lies in the rationalization of ways and means for future improvements in ultimate airplane performance through sophistication of the powerplant system. While suggestion is admittedly easier than the actual accomplishment, the development process nevertheless thrives on suggested ideas. Some feasible avenues of approach to lightweight lift engine design are suggested in the following brief discussion.

Compressor

Conventional turbojet engines are generally designed to encompass a wide range of operating conditions, such as engine speed, which is usually synonymous with thrust, altitude and flight Mach number. In order to operate with reasonable efficiency at all of these variable conditions, a number of pacifying compromises must be made in internal flow conditions and necessitates the incorporation of variable geometry. The compressor component is the most sensitive component since it has the biggest air conditioning job to do. Typical stage matching, as shown by the familiar flow/pressure-ratio

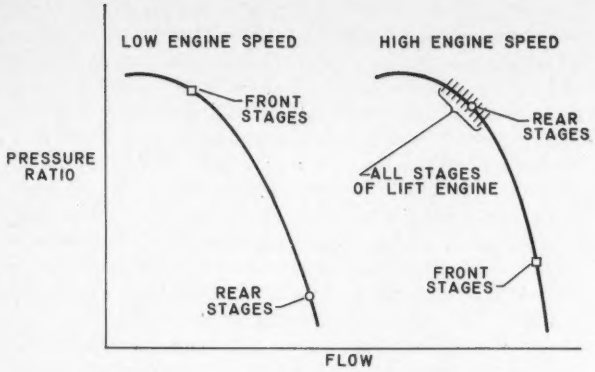

Figure 13
Typical compressor stage characteristics

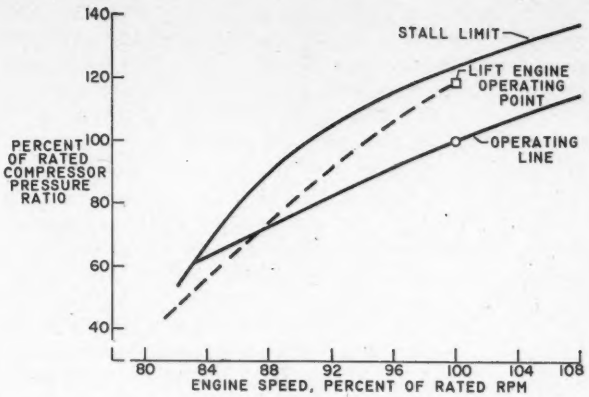

Figure 14
Typical compressor operating characteristic

relationship in Figure 13, is such that at low engine speeds the inlet compressor stages operate critically near stall while the rear stages are relatively unloaded. At high engine speed the reverse is true, with the exit stages operating critically and the inlet stages partially unloaded. This mismatching situation is generally imposed by the requirement for operation over a wide range of thrust levels and by the necessity for rapid acceleration of engine speed. The customary crutches used to maintain stable engine operation include utilization of conservative blade loading throughout the compressor and the incorporation of variable position stators and interstage air bleeds, all of which tend to increase engine specific weight.

In contrast to this, the VTOL lift engine needs only to operate effectively at essentially sea-level conditions and at a single engine speed with no special requirement for rapid acceleration. This implies that the stage matching is less critical so that each compressor stage can be loaded nearer to the individual stall limit than ordinarily practised. The overall effect is illustrated on the compressor characteristic in Figure 14, where compressor pressure ratio is shown as a function of engine speed. Ordinarily the margin between the operating and stall pressure ratio is 10% to 25% of the design operating pressure ratio. In the case of the lift engine this stall margin might conceivably be reduced to about 3% to 5% of the operating pressure ratio and the acceleration path might be as shown by the broken line. Thus, by operating the compressor close to its physical limit a given pressure ratio can be obtained with fewer stages which in turn means lower weight.

While conventional compressor blade loadings could be increased in lift engine design to achieve lower weight, another interesting approach to this end is available through use of transonic compressor blading. Research results indicate that the high inlet axial Mach numbers (0.7 and greater) and low hub/tip diameter ratios (0.3) can produce flow rates per unit frontal area that are 25% to 40% greater and stage pressure ratios 15% to 20% higher than those obtainable in subsonic compressor design practice with comparable efficiency[7]. Thus, for a given engine thrust level the higher flow capacity implies smaller engine diameter and the higher stage loading implies a decrease in the number of stages required for a given overall pressure ratio.

Thus, for lift engine design where the overall pressure ratio requirement is moderate and operation is at a fixed point, substantial reductions in engine specific weight can be realized through consideration of high blade loadings, high flow rates, reduced stall margin requirements and moderate engine speed acceleration requirement. Furthermore, since the lift engine operates at essentially zero-ram inlet conditions and low pressure ratio, alloys of lightweight metals may be adequate for compressor rotor and case construction.

Combustor

The combustor component may also be a source of weight savings in the lift engine not only in itself but also in its influence on the overall engine structure. Ordinarily, the conventional engine combustor is designed to cope with a wide range of operating conditions represented in Figure 15 by combustor pressure and combustor temperature rise. The combustor becomes necessarily long and large in diameter to provide the burning length and velocities conducive to efficient combustion at the most adverse conditions. In contrast to this, the lift engine combustor is required to operate effectively at essentially rated speed at sea level where

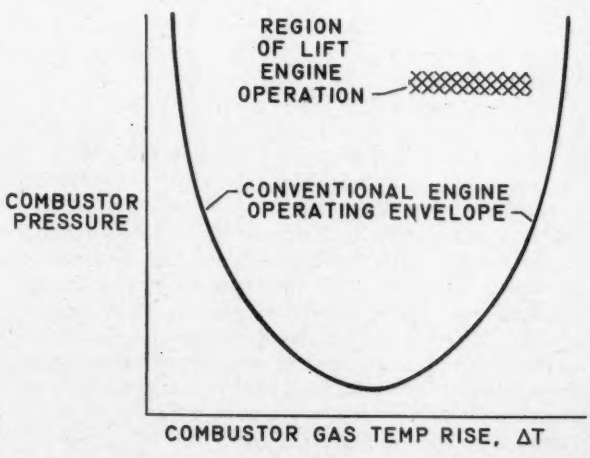
REGION OF LIFT ENGINE OPERATION
CONVENTIONAL ENGINE OPERATING ENVELOPE
COMBUSTOR PRESSURE
COMBUSTOR GAS TEMP RISE, ΔT

Figure 15
Combustor operating requirements

combustor pressure levels are sufficiently high that the combustion process can be accomplished efficiently in short, small-volume combustors. The single-point operating requirement may also allow simplifications in the fuel injection system through elimination of such refinements as duplex nozzles, double manifolds, flow dividers etc, ordinarily required for good fuel atomization at variable flight conditions.

Aside from the inherent weight savings associated with reduced volume of the combustor, the combination of the short combustor length with a smaller number of compressor stages could lead to an engine design having only two bearings rather than the customary three or four. The elimination of a bearing and its associated support structure together with a shortened turbine shaft implies appreciable weight savings.

Turbine

The lift engine turbine component does not, at least outwardly, lend itself readily to weight reduction since it must operate under the same high gas temperature, high-stress conditions as do the turbines of conventional engines. However, there are some possibilities to explore.

It would be expected that, since the compressor pressure ratio is low, a single stage turbine would be adequate. An approach to reducing the weight of this turbine stage is to see what could be done about the size or tip diameter. If it is accepted that the turbine weight is some function of the total enthalpy drop and of the square of the tip diameter, an optimization of turbine performance and weight with airplane range may permit a compromise in turbine efficiency with reduced tip speed ratio that results from reduction of tip diameter. For a given rotating speed a decrease in tip diameter reduces disk and blade stresses which in turn results in lower weight, but it also means higher blade loadings and higher turbine exit axial Mach numbers that are related to the loading limit and turbine efficiency as shown in Figure 16. In conventional design the turbine exit axial Mach number is selected to be between 0.50 and 0.60 which provides a loading margin of 20% to 35% below limiting loading to account for changes in compressor-turbine matching due to variation of component efficiencies with flight conditions and to give good acceleration characteristics.

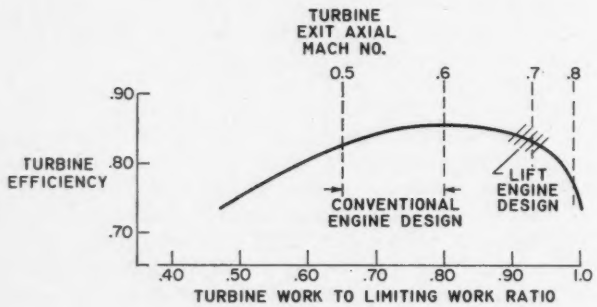

Figure 16
Effect of turbine loading on efficiency

In the case of the lift engine that operates at essentially a fixed point, the matching problem is somewhat alleviated by the consistency of component efficiencies and it therefore seems reasonable for the turbine to operate at a higher exit axial Mach number of, say, 0.70, which corresponds to a loading of 90% of the limiting load without appreciable sacrifice in efficiency. The required turbine work could thus be obtained with a turbine of smaller diameter and weight. The net gain in terms of airplane range would depend on the shrewdness of the barter between turbine weight and efficiency.

The foregoing discussion has dealt with turbines employing subsonic flow through the turbine blading. In discussing the turbine weight problem, it would be grossly amiss to neglect consideration of the transonic turbine design for lift engine application, particularly if transonic compressor staging is to be utilized. Research in this area has indicated that marked improvements in turbine size and weight can be achieved by use of transonic flow Mach numbers through the turbine. This approach permits turbine loading ratios of about 0.97 without sacrifice in turbine efficiency and also reduces the wheel stress levels by about 15% to 20% with respect to conventional turbine design.

While the potential weight reductions obtainable in turbine design are in any case small compared with that obtainable by specific design of the compressor and combustor, the endeavour for lowest engine weight should, however, be relentless on all fronts.

Exhaust system

The tailpipe and exhaust nozzle assembly should be as short as possible in the interest of conserving weight. A simple convergent nozzle would, of course, be adequate. Since precious thrust can be lost through dirty design, care should be exercised to keep the internal flow passages aerodynamically clean.

Composite lifting powerplant

The physical geometry of the lift engine would undoubtedly vary considerably from one designer to another. For a confessed lack of design ingenuity, the best that can be done here is to indicate relative size, as illustrated in Figure 17. With the combinations of a

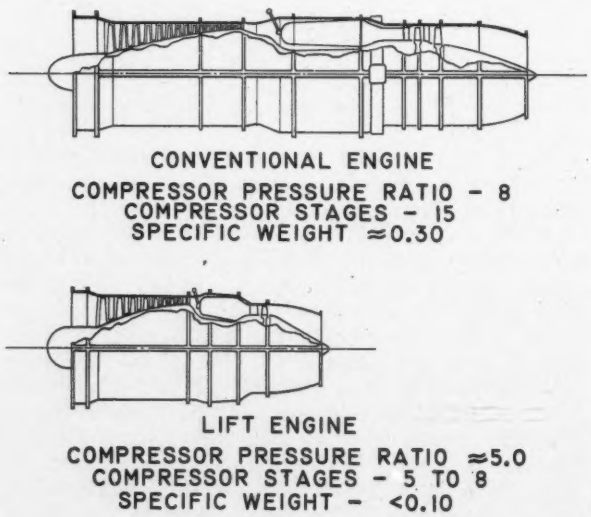

Figure 17
Comparison of engine size

highly loaded short compressor, short combustor and minimum tailpipe the lift engine might conceivably be less than half the length of a conventional turbojet engine. The diameters would be about the same for equal thrust output since the air handling capacity per square foot of frontal area would be similar. The lift engine features two bearings, a short turbine shaft, and a single-stage turbine. Utilization of transonic rotating components would, of course, result in an even shorter engine and smaller engine diameter for a given thrust level. It is conceivable that by cutting all the corners, lift engine specific weights might be achieved that would be between one-third to one-quarter that attained in conventional engine design. The thrust rating of the lift engine would apparently be compromised between the engine-out reliability of a large number of small engines and the installation simplicity of a small number of large engines. This choice is left to the discretion of the aircraft designer.

Some additional general factors that may be considered in the overall lifting powerplant installation include the low g loads on engine structure, centralized accessories such as fuel pumps, lube pumps, control systems etc. Individual engine starters may be eliminated by the use of engine turbine air-jet starters supplied with high-pressure air from the cruise engines. Factors such as these, and many others not discussed in this brief treatment of the subject, must be considered in arriving at a neat lifting powerplant package that embodies the characteristics of least weight per pound of thrust required in maximizing the performance of the turbojet-powered VTOL transport airplane.

CONCLUSION

The discussion has dealt with some of the general as well as specific turbojet powerplant-design considerations for application in a pure hoverjet VTOL transport airplane. An attempt has been made to establish overall powerplant requirements, to define desirable engine characteristics for VTOL application, and to indicate practical approaches toward improvement of engine specific weight through component design leading to better airplane performance.

The analysis indicates that the three important factors required in maximizing VTOL jet transport range/payload performance are: (1) minimum overall thrust to gross weight ratio, (2) minimum hover time and (3) minimum lift engine specific weight. The first two factors are matters of selection and compromise in the design of the aircraft to suit a particular flight plan and philosophy of operation. The third is achieved by a marriage of the engine characteristics best suited to VTOL application and the liberties that can be taken in engine design in view of unique and limited operating requirements. As a separate entity in air transportation, the range/payload performance of the turbojet-powered VTOL jet transport is of practical magnitude within the existing state of the art. Efforts directed toward improvement in turbojet lift engine weight to thrust ratio will be well rewarded in airplane performance.

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CORRIGENDUM

LETTER TO THE EDITOR

In the March 1959 issue of the Journal (p 119, Ottawa Special Meeting), it has been erroneously stated that the electric-arc driven, Hotshot wind tunnels which have been developed by ARO, Inc. at the AEDC can realize "velocity simulation up to Mach 8 and Mach number simulation up to Mach 10".

Actually, for routine testing these hypervelocity tunnels operate in the Mach number range from 10 to 20 and at velocities from 10,000 to 17,000 ft/sec. The density range corresponds to altitudes between 120,000 and 180,000 ft.

Tullahoma, Tenn. J. Lukasiewicz

COMPUTER APPLICATION TO FUEL SYSTEM DESIGN AND DEVELOPMENT†

by J. M. Tusiewicz*

Lucas-Rotax Limited

SUMMARY

The problem of turbojet engine control is briefly outlined. Possible approaches to the control solution are discussed, then one version of the fuel system is described. Dynamics of the basic element of any hydraulic control system are examined. A few typical problems are discussed and applicable computer solutions are included.

In the second part, turbojet engine dynamics are studied, both from the analytical and from the experimental point of view. Different levels of engine simulation are discussed. A more detailed description of an "economy" engine simulator is given. A short examination of the significance of the computer contribution to fuel system design and development concludes the paper.

INTRODUCTION

THE object of the fuel system engineer is to ensure that, under all conditions, just the right amount of fuel is passed to the engine burners. The object of the engine designer is to ensure that the complexity of this task increases very rapidly. When one looks back at some early controls and compares them with a contemporary fuel system of an advanced engine, it becomes evident that the turbojet engineer is achieving his goal with an extraordinary measure of success.

SIMPLE ENGINE SYSTEM

Early engines had a considerable degree of simplicity. A single rotor engine with a fixed nozzle is shown diagrammatically in Figure 1. Various stations are specified by number: 0 environment, 1 intake, 2 compressor delivery, 3 combustor outlet, 4 turbine outlet, 5 exhaust nozzle. These numbers, when used as subscripts to an engine variable, indicate the position in the engine at which the variable is being considered. A block diagram of the same engine is shown in Figure 2. There is only one direct input to the engine: the fuel flow W. There are also two indirect inputs: the intake pressure P_1 and intake temperature T_1. These two may be controlled indirectly by changes in aircraft speed and altitude. Two outputs which are of interest appear in the diagram. Obviously, the pressures and temperatures anywhere within the gas stream could be included among the outputs.

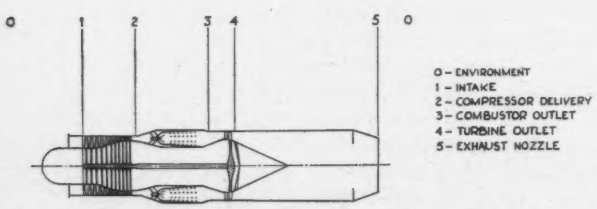


Figure 1
Simple turbojet engine schematic diagram

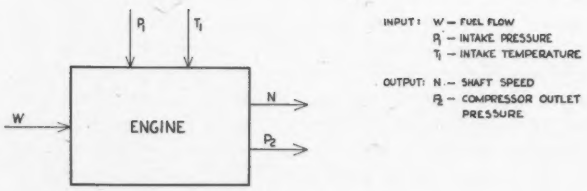


Figure 2
Simple turbojet engine block diagram

The function of the control is to provide at any power level setting a fixed percentage of the thrust available under given conditions. Thus, at full throttle 100% of the thrust available under given conditions should be obtained. This 100% depends on the engine structural strength under operating conditions. One of the most obvious limitations is the rotor speed. As a fair approximation:

$$W/P_1\sqrt{T_1} = a + b(N/\sqrt{T_1})^3 \qquad (1)$$

where a and b are constants for a given engine. It follows the fuel flow at full throttle and for a fixed maximum speed is:

$$W = P_1(a\sqrt{T_1} + b\,N^3_{max}/T_1) \qquad (2)$$

Hence the fuel flow required at full throttle is proportional to the intake pressure, but it is also a function of the intake temperature.

†Paper read at the Annual General Meeting of the C.A.I. in Toronto on the 27th May, 1958.
*Chief Performance Engineer.

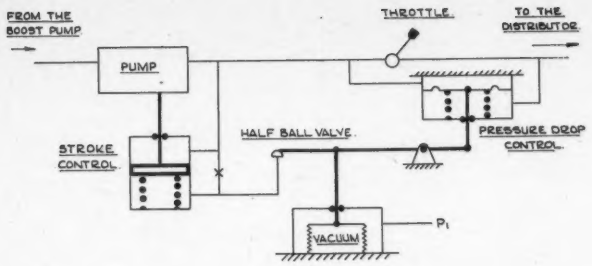


Figure 3
Simple flow control

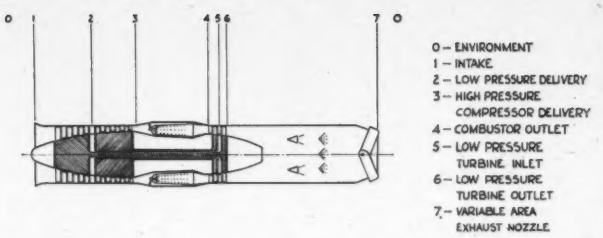

0 1 2 3 4 5 6 7 0
0 – ENVIRONMENT
1 – INTAKE
2 – LOW PRESSURE DELIVERY
3 – HIGH PRESSURE COMPRESSOR DELIVERY
4 – COMBUSTOR OUTLET
5 – LOW PRESSURE TURBINE INLET
6 – LOW PRESSURE TURBINE OUTLET
7 – VARIABLE AREA EXHAUST NOZZLE

Figure 5
Modern turbojet engine schematic diagram

The intake pressure is easily measured and can be applied for control purposes without difficulty, but not the intake temperature. The temperature sensing elements have always been slow and not easily adaptable for control purposes. Fortunately, in the early days when the aircraft speeds were relatively slow, ram and seasonal temperature changes could be neglected and a simple relationship between P_1 and T_1 (ICAN standard atmosphere) gave satisfactory approximation. In this manner it was possible to schedule the fuel flow at full throttle as a function of intake pressure only. The thrust is approximately proportional to the fuel flow; hence with the intake pressure constant the fuel flow selected at the intermediate throttle angles had to be proportional to the power lever position.

Right from the beginning the British developments went in the direction of a variable stroke pump and a scheduling system. A simple control of this type is shown in Figure 3. P_1 and the pressure drop across the throttle are balanced on the lever. An out-of-balance force produces displacement of the half ball valve, which in turn reduces (or increases, as the case may be) the pressure on the spring side of the pump stroking piston. The change of the stroke is in the direction of restoring the balance of forces on the lever. Thus at any time the pressure drop across the throttle is proportional to the engine intake pressure. By making the throttle area a function of the power lever angle, a suitable relationship between the power lever position and the fuel flow can be obtained. For overspeed protection a centrifugal pressure generated by radial drillings in the pump rotor would operate on the same stroke control when the preset speed was reached. The schematic diagram of a pump with the governor is shown in Figure 4.

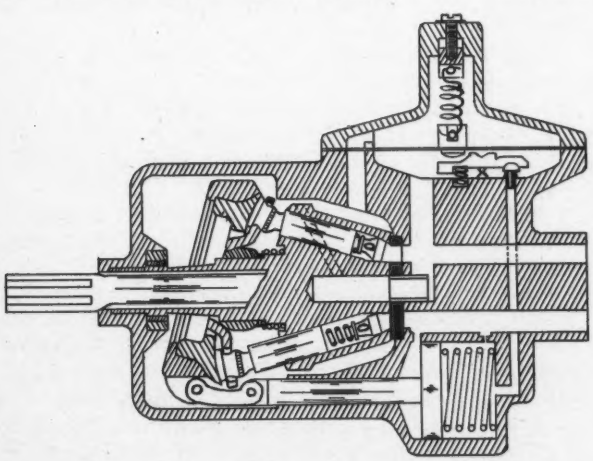


Figure 4
Lucas-Rotax "C" size fuel pump schematic diagram

The American development has been in the direction of a gear pump and an all-speed governor system. At times there was considerable discussion concerning the relative merits of the two approaches. Although the passing years seem to have removed many a point of contention, the old arguments are still heard occasionally. Far from being 100% biased, this writer would like to put on record:

(1) The scheduling system still provides protection in case of a governor failure.

(2) On some advanced engines a completely satisfactory schedule is not practical and an approximate scheduling must be combined with part or full range governing.

(3) There are many aspects of an advanced engine which override the speed throttle relationship. It makes no difference to the complexity of the system whether these overrides are applied on the schedule or whether they operate on the speed selector.

(4) Today, there is hardly an all-speed control that would not include some form of scheduling. Conversely, most schedule controls are combined with at least part-range governing. The difference seems to be in the name only.

ADVANCED ENGINE SYSTEMS

As higher and faster flying aircraft were required, the engines grew more complex and so did their controls. A schematic diagram of a modern turbojet engine is given in Figure 5. This example shows a twin rotor engine with an afterburner and a variable exhaust nozzle. Other configurations are of course possible. The block diagram for this engine is presented in Figure 6.

There are three direct inputs to the engine:

(a) the engine fuel flow,

(b) the afterburner fuel flow, and

(c) the nozzle area.

With other applicable inputs, like the variable intake, variable guide vanes and compressor bleed, the list of possibilities is growing rapidly. There are also more outputs to control e.g.:

N_L	Low pressure rotor speed
N_H	High pressure rotor speed
P_3	High pressure compressor delivery pressure
T_7	Exhaust nozzle temperature

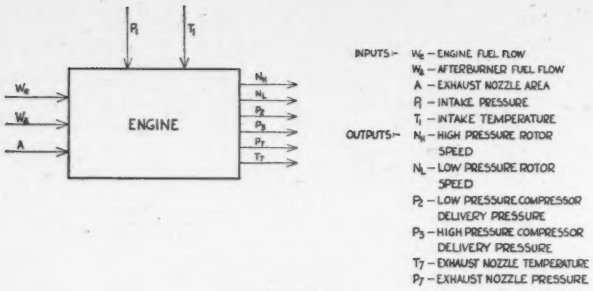

ENGINE
INPUTS:- W_e – ENGINE FUEL FLOW
W_a – AFTERBURNER FUEL FLOW
A – EXHAUST NOZZLE AREA
P_1 – INTAKE PRESSURE
T_1 – INTAKE TEMPERATURE
OUTPUTS:- N_H – HIGH PRESSURE ROTOR SPEED
N_L – LOW PRESSURE ROTOR SPEED
P_2 – LOW PRESSURE COMPRESSOR DELIVERY PRESSURE
P_3 – HIGH PRESSURE COMPRESSOR DELIVERY PRESSURE
T_7 – EXHAUST NOZZLE TEMPERATURE
P_7 – EXHAUST NOZZLE PRESSURE

Figure 6
Modern turbojet engine block diagram

For the simple engine of Figure 1, limiting the engine speed gave an acceptable assurance of all pressures and temperatures remaining within their limits. No more so with the engine of Figure 5; e.g. by changing the nozzle area and afterburner fuel flow a wide range of nozzle temperatures is possible with a constant engine speed. In order to extract the maximum performance from the engine, some engine outputs have to be controlled selectively depending on the intake conditions and on other engine outputs.

One configuration of a modern fuel system is shown in Figure 7. The heart of the system is the main metering orifice. The area of this orifice is some function of the compressor delivery pressure and/or intake pressure. The bellows force is balanced on the lever by the spring pull. The higher the pressure the more extended the spring. The spring controls a shaped plunger within the metering orifice. The position of the plunger determines the metering orifice area. The pressure drop across the metering orifice is controlled by the power lever. Moving the power lever forward increases the area *a* and, hence, the pressure on the spring side of the diaphragm rises, closing the orifice *c*. The pressure behind the servo throttle valve piston drops and the piston moves to increase the servo throttle valve area until the selected pressure drop is obtained and the equilibrium of forces on the diaphragm is restored. The system is particularly adaptable for applying schedule overriding signals, e.g. opening an orifice in parallel with orifice *a* increases the flow above the schedule, while opening an orifice in parallel with orifice *b* reduces the fuel flow below the scheduled value.

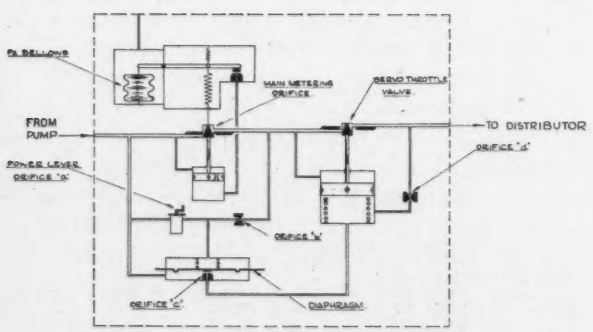


Figure 7
Modern fuel system schematic diagram

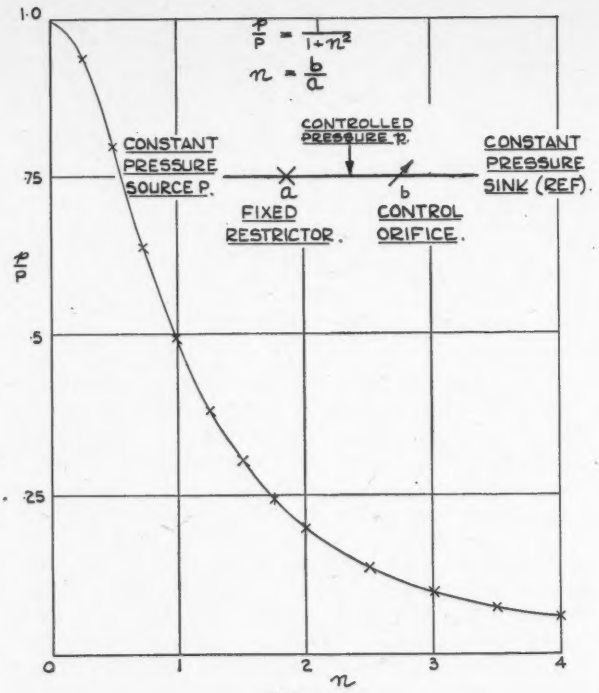


Figure 8
Hydraulic potentiometer

Control elements

A pump and an orifice are the essential parts of any fuel system. Even the most sophisticated electronic control would not be possible without them. The orifice law is

$$W = ka\sqrt{P} \qquad (3)$$

where k – flow constant
a – orifice area
P – supply pressure

A multiplier, which is also a linear or square-rooting device, is employed by the hydraulic system in all three capacities. In fact, as shown in Figure 7, a hydraulic fuel system is little more than a conglomeration of fixed and variable orifices.

The basic control function is provided, in such a system, by a hydraulic potentiometer – a pair of restrictors interposed between the high pressure source and low pressure sink. When one or both restrictors vary, the intermediate pressure changes as a function of their area ratio. This relationship is shown in Figure 8.

A fairly common form of the variable restrictor is a half ball valve. This may be positioned by the sensing element of the control. Generally, it can be used with a low power level signal. If a hydraulic jack is connected to the intermediate pressure of the potentiometer, the low level input signal may produce an output at a higher power level. The motion of the hydraulic jack is used, in turn, to control a valve, to stroke a piston pump etc.

A simple hydraulic actuator is shown in Figure 9. It is assumed that the load is proportional to the displacement X, the inertia of moving parts is negligible, the fluid is incompressible, and there is no leakage. Under

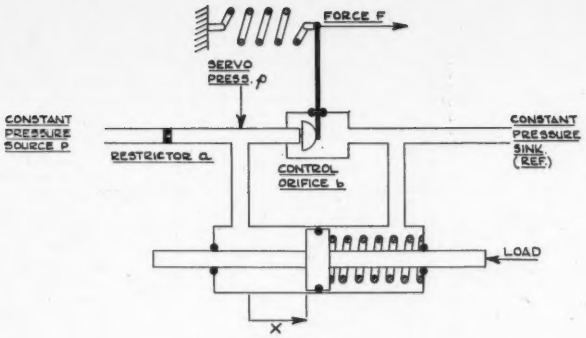


Figure 9
Simple hydraulic actuator, open loop

these conditions the equation of flows and the equation of forces may be combined into:

$$\frac{dp}{dt} = \frac{kaR}{A^2}(\sqrt{P-p} - n\sqrt{p}) \tag{4}$$

where R – spring rate
A – piston area
p – servo pressure
n – orifice area ratio

This is a first order, non-linear, differential equation with parametric forcing. With moderate effort it may be integrated to give time t taken to reach any servo pressure p:

$$t = \frac{2\gamma_2 A^2\sqrt{P}}{kaR}\left\{\left[\log\tan\left(\frac{\pi}{4}+\frac{\theta}{2}\right) - \sin\theta\right]\Bigg|_{\sin\theta=\sqrt{\frac{P-p}{P-p_2}}}^{\sin\theta=\sqrt{\frac{P-p_1}{P-p_2}}} + \left[\frac{1}{2}\log\left(\frac{y+1}{y-1}\right) - y\right]\Bigg|_{y=\sqrt{\frac{p}{p_2}}}^{y=\sqrt{\frac{p_1}{p_2}}}\right\} \tag{5}$$

$$\gamma_2 = \frac{p_2}{P}\sqrt{1-\frac{p_2}{P}}$$

where p_1 – initial servo pressure
p_2 – final servo pressure

This solution applies to the step increase of the controlling orifice area.

It would not be fair to say that this solution is entirely useless, yet even at the first glance it is obvious that its application to the general system analysis must be very limited.

Eq. (4) may be linearized before attempting a solution. Luckily, flow, Eq. (3) (from which Eq. (4) derives), is nearly linear with respect to the pressure drop, for up to ± 30% pressure drop change the linearized flow remains within ± 1% from the exact value (Figure 10).

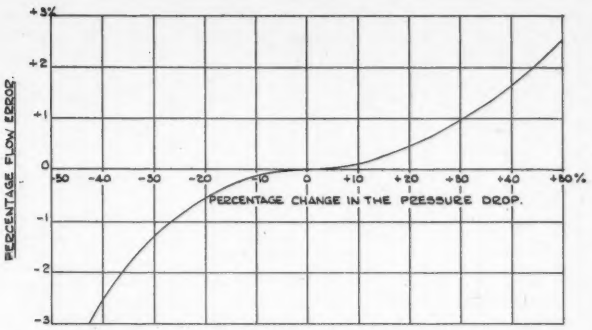


Figure 10
Linearized flow through an orifice
$W \propto p$

The linearized incremental version of Eq. (4) is:

$$T\frac{d(\Delta p)}{dt} + (\Delta p) = -B(\Delta n) \tag{6}$$

$$T = \frac{2\gamma_1 A^2\sqrt{P}}{kaR} \qquad B = 2P\sqrt{a_1}\,\gamma_1$$

$$a_1 = p_1/P \qquad \gamma = a_1\sqrt{1-a_1}$$

The step response to (Δn) follows:

$$\Delta p = B(e^{-t/T} - 1)(\Delta n) \tag{7}$$

It may be of interest to note that the "local" response is the slowest when the power transfer is at the maximum, i.e. when the servo pressure is ⅔ of the source pressure. The maximum time constant is then:

$$T_{max} = \frac{.77A^2\sqrt{P}}{kaR}$$

Since A^2/R is proportional to the volume displaced divided by the source pressure, it follows that the time constant is proportional to the volume displaced and inversely proportional to the product of the restrictor area and the square root of the source pressure.

An efficient hydraulic actuator of this simple type would have probably a preload of some $0.4\,PA$, maximum load of $0.875\,PA$ and a spring rate of $0.475\,PA/$ stroke. Under these conditions the local time constant would always be more than $0.0124\,V/a\sqrt{P}$ and less than $0.0154\,V/a\sqrt{P}$.

The average value would be good enough for most applications.

This approach may be extended to other configurations of hydraulic actuators and to other loads. Factors like leakage or inertia are easily included. The number of various possibilities is enormous.

To provide an example for the simple actuator of Figure 9:

source pressure	P	100 psi
piston area	A	1 sq inch
restrictor	a	0.001 sq inch
effective rate	R	100 lb/inch
stroke	X_{max}	0.475 inch
flow constant	k	100 sq in/sec $\sqrt{\text{lb}}$

When the area ratio n changes from 0.5 to 1

$$\Delta p = 32(e^{-1.4t} - 1) \tag{8}$$

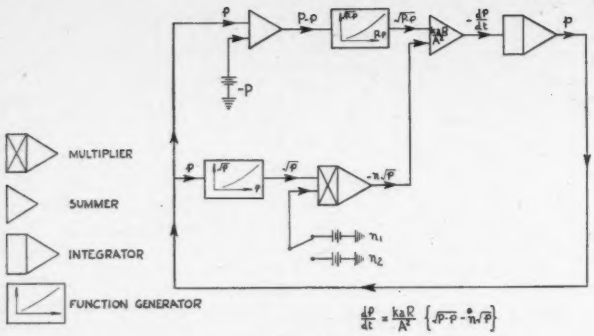


Figure 11
Computer set up for the actuator of Figure 9

All this is straightforward if not quite exact. An accurate solution may be obtained from Eq. (5), or by setting Eq. (4) on an analogue computer. One possible set-up is shown in Figure 11.

For comparison, Solution (8) and the computer solution have been recorded side by side in Figure 12a. Although the step is relatively large the agreement is quite good; Transfer Function (6) is useful.

The restrictor area ratio is but one possible input. The response to the source pressure change at constant n may be described:

$$T\frac{d(\Delta p)}{dt} + (\Delta p) = \alpha_1(\Delta P) \tag{9}$$

Again the step change response (step in P from 100 psi to 200 psi, $n = 0.5$) is compared with the computer solution (Figure 12b). The result is acceptable.

However, these two changes could take place simultaneously. The linearized equation would be then:

$$T\frac{d(\Delta p)}{dt} + (\Delta p) = \alpha_1(\Delta P) - B(\Delta n) \tag{10}$$

A simultaneous step in P from 100 psi to 200 psi and n from 0.5 to 1.0 is considered. Again the computer solution and the linearized solution are recorded together (Figure 12c). The latter is wildly off the mark.

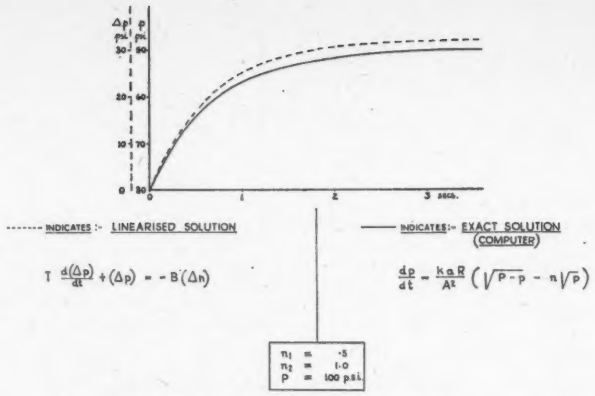
Eq. (10) is not good enough except for very small departures from the equilibrium.

The arrangement of Figure 9 is an open loop system. For the same force F the jack may take various positions depending on the changing load conditions. From this point of view the closed loop system of Figure 13 is much more useful. Neglecting the slight angular changes in the position of the lever the input force is always balanced by the pull of the piston spring and, hence, the jack position X is a function of force F only. Also the speed of response is improved. The closed loop system is represented by the block diagram of Figure 14b. The transfer function is:

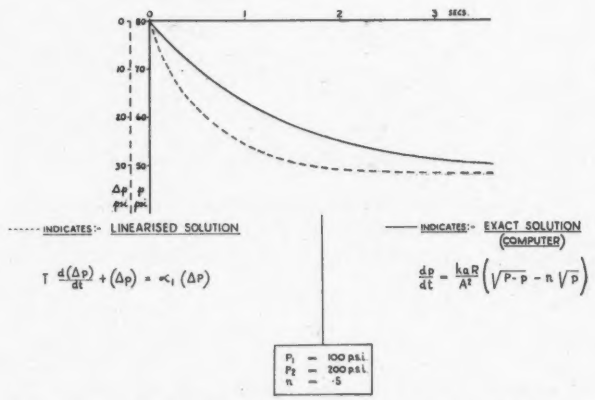
$$\frac{\Delta X}{\Delta F} = -\frac{BK_1K_2}{(1 + BK_2)}\frac{1}{(T_1 s + 1)} \tag{11}$$

where

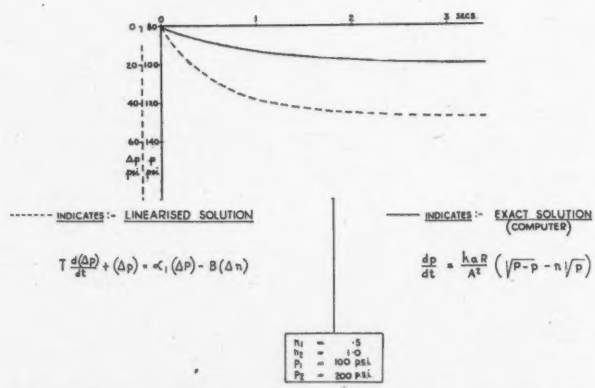
$$T_1 = \frac{T}{1 + BK_2}$$



(a) Response of the simple actuator to a step increase of the controlling orifice area



(b) Response of the simple actuator to a step increase of the supply pressure P



(c) Response of the simple actuator to simultaneous step increases of the supply pressure and controlling orifice

Figure 12

Thus, for a given configuration of the actuator the closed loop time constant is smaller than the open loop one. The amount of reduction depends on the loop gain BK_2. The response is faster because the initial half ball displacement goes beyond the final equilibrium position. As a rule, even with a relatively small input, the area ratio saturates initially, the control restrictor opens (or

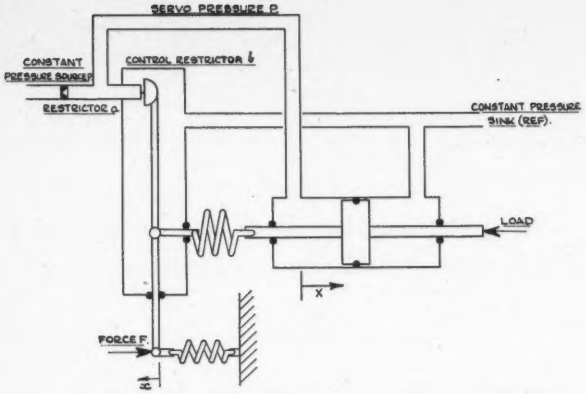


Figure 13
Simple hydraulic actuator, closed loop

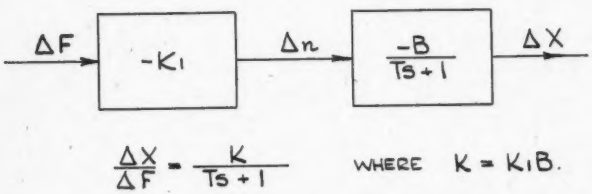
closes) fully. It remains open (or closed) until the changing piston spring pull brings the half ball again within the effective control range. There are clearly two modes of operation here:

(a) With the half ball fully open (or closed) response occurs at almost constant rate

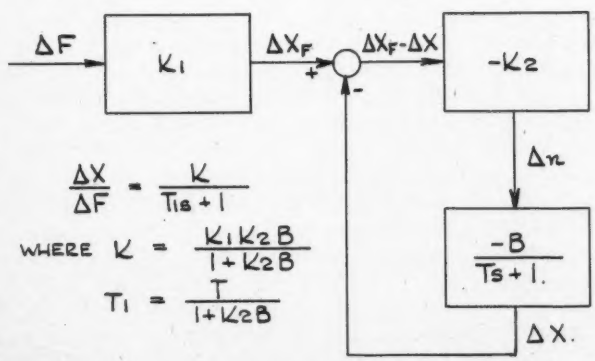
$$\frac{dX}{dt} = -\frac{AB(\Delta n)}{RT} = -\frac{ka\sqrt{p_1}(\Delta n)}{A} \qquad (12)$$

where (Δn) is the maximum change possible, i.e. from the initial equilibrium ratio n_1 to fully open or closed, as the case may be. The major part of the response occurs under the constant rate conditions.

(b) Within the control range Eq. (11) holds good and a short duration adjustment of the final equilibrium position take place. The previous example is extended to cover the closed loop case:



(a) Actuator in the open loop system of Figure 9



(b) Simple actuator in the closed loop system of Figure 13
Figure 14

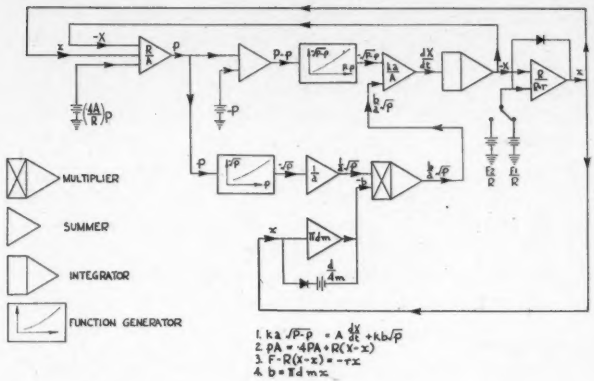


Figure 15
Computer set up for the actuator of Figure 13

effective arm, force F		1 inch
effective arm, piston spring		1 inch
effective arm, half ball	(m)	2 inches
controlling orifice diameter	(d)	0.064 inch
trim spring rate	(r)	20 lb/inch

Figure 15 shows a possible analogue arrangement for this system.

Two solutions are recorded in Figure 16:

(a) Force F suddenly reduced from 40 lb to 20 lb.
(b) Force F suddenly increased from 20 lb to 40 lb.

By now the relatively trivial problem of the simple element dynamics has been belaboured considerably. It has been demonstrated that the hydraulic control elements are subject to various degrees of non-linearity. The flow through an orifice is mildly non-linear with respect to the pressure drop across it. In most cases the linearized relationship does provide an analytically useful substitute. On the other hand, the two modes of operation of the hydraulic actuator of Figure 13 constitute an essential non-linearity which is difficult to describe usefully in linear terms. As for any other problem the lengthy and laborious numerical analysis may be applied. It will lead to a numerical series which yields, after plotting, the response to a particular input. The relationship between the system parameters and the obtained result will be completely obscured. An analogue computer provides here by far the better avenue of approach. In the analogue set-up all the system parameters are physically simulated and easily adjustable. The actual responses may be observed directly and re-

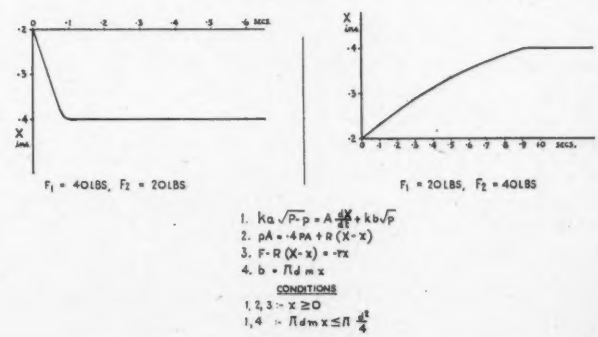


Figure 16
Response of the actuator of Figure 13

peatedly while 'areas', 'spring rates', 'pressures', 'forces' etc are being changed at will. Moreover, with proper servo outputs the analogue computer provides a test facility for the actual hardware, mixed possibly with some components which have not reached the drawing board stage.

After the individual control elements have been investigated, they are combined into a fuel system and the investigation is continued. The proper measure of the system performance, however, may be obtained only by evaluating the dynamics of the system combined with the engine for which it is intended.

Engine dynamics

Dynamic characteristics of the engine may be determined either analytically or experimentally. The engine speed response to a small change in fuel flow approximates closely to an exponential lag. This lag is due, obviously, to the rotor inertia. Whenever the fuel flow is increased above the value required at the existing speed an excess torque is being applied to the rotor in effect, and the rotor accelerates until the new equilibrium is reached.

At any time

$$\frac{I}{g}\left(\frac{2\pi}{60}\right)\frac{dN}{dt} = \text{excess torque} \tag{13}$$

where I — total inertia of rotating masses

It has been shown by various writers (e.g. Reference 1) that Eq. (13) may be reduced with a good approximation to

$$\frac{dN}{dt} \propto \frac{(1 - T_4/T_3)(\Delta W)}{IN} \tag{14}$$

The engine speed response to a small step change in fuel flow as described in terms of an exponential lag is

$$N = N_o + \Delta N(1 - e^{-t/T}) \tag{15}$$

It follows, that the rate of change is

$$\frac{dN}{dt} = \frac{\Delta N}{T} e^{-t/T} \tag{16}$$

and in the first instance

$$\frac{dN}{dt} = \frac{\Delta N}{T} \tag{17}$$

Hence, combining Eqs. (17) and (14) the local engine time constant at any given speed is obtained

$$T \propto \frac{IN}{1 - T_4/T_3}\frac{dN}{dW} \tag{18}$$

At sea level static conditions, for many engines, the constant of Eq. (18) is around 8×10^{-8} when

I	rotor inertia in lb ft^2
N	rotor speed in rpm
dN/dW	engine gain rpm/pph
T_4/T_3	turbine temperature ratio

The engine time constant at various flight conditions may be obtained from the relationship

$$T P_1/\sqrt{T_1} = \text{constant} \tag{19}$$

once the time constant at any one set of conditions is known.

Eq. (18) gives an approximate but useful value of the engine time constant. A local engine time constant and a corresponding local engine gain describe fully dynamic characteristics of the engine at one particular operating point within the assumptions made. Elaborate methods can be employed to derive the engine time constant by more rigorous steps — the "exact" results do not seem to justify the effort.

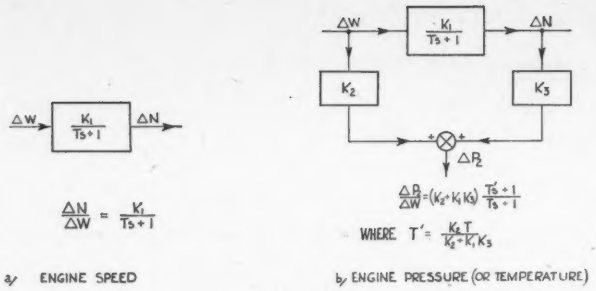


Figure 17
Engine transfer functions

The dynamic characteristics of the engine may also be obtained experimentally. The local gain at any speed is the slope at that point of the steady state engine speed curve plotted against the fuel flow. The engine time constant may be measured by recording the engine speed change, either in response to a small step change in the fuel flow or in response to small oscillations of the fuel flow at various frequencies. The former method is relatively fast and does not require excessive engine time. On the other hand a good fuel flow step change is difficult to produce. Also, the numerical result depends on the slope measurement from the recording. The slope measurement error is likely to be appreciable.

The frequency response method is laborious and requires much more engine time. The recording at any one engine speed must involve some 20 runs at different frequencies of oscillations.

In order to provide undistorted and readable traces at frequencies up to some 10 cps the transducers and the recording equipment must be fast, sensitive to the signal and insensitive to the noise and vibration.

The amplitude ratio and the phase shift between the fuel flow and the engine speed vary at different frequencies of the fuel flow input. Their variations give precise information about the engine dynamics.

Both these experimental methods confirm that the engine speed response to the fuel flow is dominated by a single time constant. However, other secondary factors are present. These factors are particularly evident in the results of a good frequency response test.

The response of other engine outputs like pressures and temperatures depends, to a large extent, on the speed response. It may be assumed that, as a fair approximation, pressure or temperature changes combine two terms — one in phase with the fuel flow and the other in phase with the engine speed.

The local transfer functions of the engine are shown in Figure 17.

Engine simulation

The basic purpose of the engine simulator is to sense the input signals, either real or simulated or both, and to generate simulated and real outputs which closely

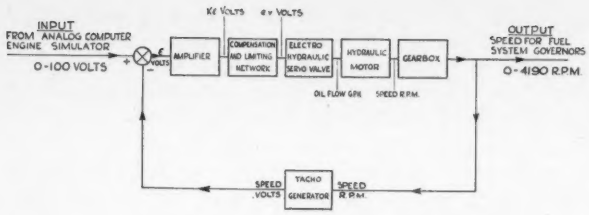


Figure 18
Control for the simulation of engine speed

represent the engine variables under the corresponding conditions.

The degree of complexity depends on the number and character of the inputs and outputs required, and on the simulation level chosen. As an example, a simple engine may be considered. For any given intake conditions there is one input only, the fuel flow. It might be required to control three outputs: engine speed N, compressor delivery P_2 and jet pipe temperature T_4. The fuel system in question may be using a mechanical range governor, a pneumatic pressure limiter and an electronic temperature limiter. Accordingly, the simulator must measure the fuel flow from the system, and convert this information into the shaft rotation (corresponding to the engine speed), the pressure (corresponding to P_2), and the voltage (corresponding to the thermocouple signal). The physical variables will be produced by various servos controlled by the computer outputs. The computer outputs are derived by solving various equations involving fuel flow and other engine parameters. At one extreme, force and flow equations for the compressor and turbine blades and similar fundamental equations for other components could be used as the starting point. At the other extreme, a single transfer function applying at a particular engine speed could be employed for each engine output. In between, there are a number of possibilities with the direct simulation of the major engine components (compressor, combustor, turbine and exhaust nozzle) probably more convenient than the others. Various approaches to the simulation problem have their advantages and disadvantages. The first principles approach would satisfy a rigorous theoretician but the cost, effort and lack of basic information make it quite unrealistic. The major components approach has cost General Electric Company three million dollars and taken seven years to complete[2]. They produced a flexible tool which will be adaptable for the simulation of engines several years ahead. One particular engine speed approach hardly justifies the effort necessary in building the output servos. A compromise solution goes in the direction of a speed variable transfer function which provides for an acceptable simulation of throttle bursts at a relatively modest cost. A simulation of this kind would require fairly extensive modifications in order to adapt it for the type of engine other than that for which it was designed. However, it can be designed to simulate complex engines. The main advantage lies in avoiding the slow and costly two variable function generators. The output servos are conventional.

The schematic diagram of the speed servo is shown in Figure 18. The actual shaft speed is measured by a DC tacho generator and the signal is compared with the computed speed signal. The difference between the

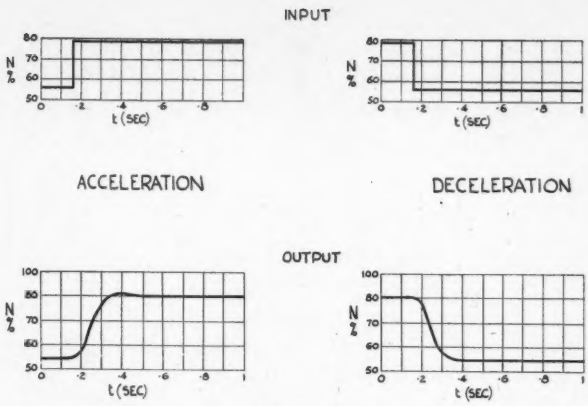


Figure 19
Typical response curves for the speed control

two signals, amplified and compensated for the servo lags, operates on an electro-hydraulic valve which in turn drives a hydraulic motor. A recording of the motor response to a step computer command is shown in Figure 19. It takes approximately 0.1 second for some 60% speed change.

The computer set-up for the speed simulation is given in Figure 20. The relationship between the engine speed and the fuel flow is

$$T(N)\frac{d(\Delta N)}{dt} + \Delta N = K(N)\,\Delta W$$

$$T(N)\frac{dN}{dt} + (N - N_o) = N_{ss} - N_o \qquad (20)$$

$$T(N)\frac{dN}{dt} = N_{ss} - N$$

It follows

$$\frac{dN}{dt} = \frac{N_{ss} - N}{T(N)} \qquad (21)$$

The time constant as a function of the engine speed is either measured on the test bed or calculated. This requires one single variable function generator. N_{ss} is the engine speed corresponding to the fuel flow measured at any instant. It accounts for another function generator. N_o is the initial engine speed and N the actual engine speed. The difference between the steady state speed

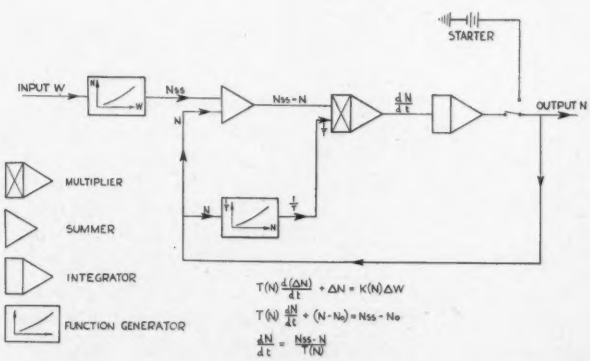


Figure 20
Computer set up for engine speed simulation

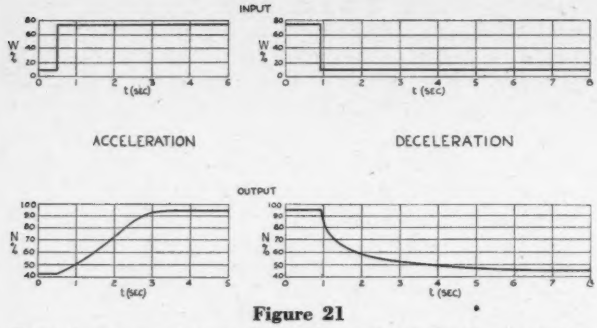


Figure 21

Typical response curves for the computer simulated engine

and the actual speed multiplied by the reciprocal of the time constant corresponding to the actual speed gives the instantaneous rate of change in speed. The response of the computed speed to a step change in the signal representing the fuel flow is given in Figure 21. It shows an acceleration of the ideal engine not affected by compressor surge. In practice, the acceleration would have to be slower to avoid the surge region. The acceleration problem is a chapter on its own and will not be discussed here. It should be stated, however, that the surge characteristic can be built into the simulator so that the engine stalls when surge conditions occur. To add an unintentional touch of reality the engine will not respond to a mere fuel flow signal when below a certain speed. A cranking speed signal has to be applied for starting, as shown in Figure 20. This, of course, is due to the infinite time constant at zero engine speed.

A computer set-up for P_2 simulation is given in Figure 22. The output signal contains two terms. The actual engine speed signal is used to compute the steady state P_2 corresponding to that instantaneous speed — this gives the steady state calibration. Further, the difference between the actual fuel flow and the equilibrium fuel flow corresponding to the instantaneous engine speed is computed. A product of this difference and of a speed function produce the other term. It adds to the output signal when the flow is excessive and subtracts when the flow is below the equilibrium value. As before, the speed functions involved are derived either analytically or from the engine test. Any other pressure or temperature may be simulated along the same lines. A

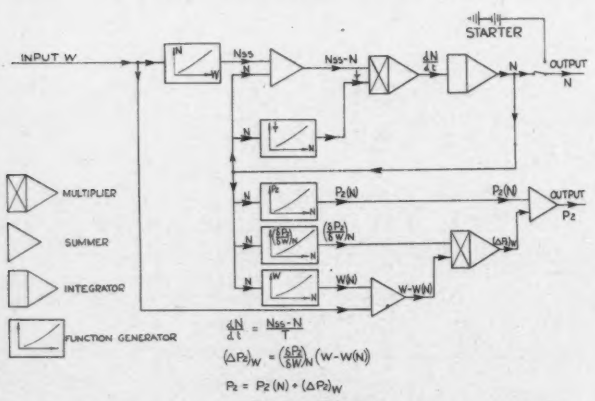


Figure 22

Computer set up for engine pressure simulation

pneumatic or hydraulic potentiometer is generally used for a pressure servo. The variable orifice is controlled by the error signal, amplified and compensated for servo lags. The error is the difference between the measured pressure and the computed pressure.

In case of a temperature output usually no servo is required; the computer signal is attenuated and passed through an RC network to simulate the thermocouple lag. In turn, the servo outputs are applied to the fuel system sensing elements and so the loop is closed.

The final check of the simulator is done by running frequency response tests at various speeds and comparing the results with those obtained on the actual engine. If these agree closely, the chances are good that the control system and simulator combination will behave in most cases as the control system and engine combination would.

COMPUTER CONTRIBUTION

The computer can make a very significant contribution towards the solution of a control problem. It offers many advantages:

(1) It speeds up laborious solutions of non-linear equations.
(2) It produces solutions to mathematical problems which were impossible to solve for practical reasons.
(3) It enables the control engineer to test his control without a very costly engine test facility.
(4) The running cost is insignificant in comparison with the cost of the actual engine time.
(5) It enables one to run an altitude engine test firmly on the ground and at a small fraction of the cost of a proper altitude test facility.
(6) It can be used to investigate a control system before it is actually made, or when it is partly completed.
(7) It can be used together with the hardware to examine the effect of changing a component or components without the necessity of making additional bits and pieces.
(8) It permits testing a control on an engine when either or both are at the proposal stage.

The computer results can be occasionally misleading. Obviously, they are only as good as the information that the computer has received. Also, there is always a point beyond which a closer simulation is not practical. With care and effort the computer can become a versatile and valuable tool in the control design and development but, in the end, the final proof of a control is on the engine.

ACKNOWLEDGMENTS

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HYDRAULICS FOR A HIGH PERFORMANCE AIRCRAFT — WHY AND HOW†

by D. Royston*

Avro Aircraft Limited

INTRODUCTION

THE object of this paper is to follow through in a general manner the steps involved in choosing the optimum power supply for the flight controls and utility services of a high performance, supersonic, interceptor aircraft.

The reasons for the choice of hydraulics are stated; followed by a discussion of some of the problems that face the hydraulic systems engineer and the solutions used to overcome them.

The paper concludes with a summary of the future of hydraulics in high performance applications.

REQUIREMENTS

Since World War II, numerous articles have been written praising the relative merits of hydraulics, pneumatics and electrics for powering actuating systems. The fact remains that the decision on which system is the best, for a particular application, can only be determined after a careful study has been made of the system requirements. In some cases the fundamental requirements dictate the use of a particular type of power supply; in others, a more detailed analysis is necessary – based on comparative weight, complexity, reliability, safety, performance, and cost of hydraulics, pneumatics or electrics – before the final choice is made. (No mention is made of the possible use of mechanical transmissions for these applications because mechanical power transmissions could not compete with the other types with respect to the various parameters considered.)

In the initial stages of scheming a new project, it is possible to make a list of controls which require some form of actuation. Examples which are applicable to a high performance aircraft are:

(1) Flight controls
(2) Speed brakes
(3) Wheel brakes – with or without anti-skid control
(4) Landing gear actuation – legs and doors
(5) Nose wheel steering
(6) Missile extension mechanisms

The flight control system

It is customary to split these up into two basic categories, flight control systems and utility systems.

In the controls listed above, (1) falls into the flight control system category; (2) sometimes does as well, depending upon the use of the speed brakes for aircraft control.

In this paper, (1) only is put into the flight control system category; the remaining have been lumped together into the utility system, which can also contain various other specialized controls, determined by the technical role of the aircraft.

To quote some realistic figures to clarify the problems involved, I am going to discuss some of the requirements for the design of these systems for the Avro Arrow, a supersonic, two place, twin engine, all weather interceptor aircraft.

The supersonic role of the aircraft dictates the need for fully powered irreversible flight controls. The power analysis of this system shows peak horsepowers of the order of 185, with an average continuous power requirement of approximately 20 hp. Combined with this are requirements for maximum control surface hinge moments in the order of 60,000 ft lb per surface. Electrics are immediately ruled out by virtue of the weight and size of electrical actuators to provide these hinge moments plus the fast control responses required. Space limitations for the control surface actuators and the high response rates required dictate the use of a system pressure of 4,000 psi, to obtain the required hinge moments.

With these fundamentals of

(1) peak powers of 185 hp,
(2) an average power consumption of 20 hp,
(3) a nominal operating pressure of 4,000 psi, and
(4) high rates of response and high circuit stiffness,

it is possible to say immediately that the only system to satisfy all these is hydraulics.

The next step is to lay out the basic systems for actuating the various services, starting with the requirement that the flight controls be powered by hydraulics.

Of prime importance here is the necessity of ensuring maximum reliability of the flight control system. The controls are to be fully powered, and manual reversion

†Paper read at the Joint I.A.S./C.A.I. Meeting in Ottawa on the 8th October, 1958.
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is not a possible method of emergency control as the forces and system response requirements are beyond the capability of direct pilot input. Duplicated hydraulic power supply is an absolute necessity.

The first layout that comes to mind is that of using one hydraulic system for normal control and a standby system, which could be switched in, in case of a failure in the normal supply system.

In a high performance aircraft which demands high control response rates, a pilot operated switching device to bring the secondary power system into operation could be too slow; therefore, an automatic device is necessary. This device is operated by the sensing of a drop in pressure in the normal supply system. A sophisticated shuttle valve can be designed to do this job. This valve would be mounted right on the actuator and duplicate pressure and return lines would be run to it; one set from the normal, and one from the standby or emergency system. To allow for rapid cut-in of the standby system, it must be kept ready to take over the instant power is demanded from it.

Using this set of ground rules, two separate hydraulic systems could be used; one for flight controls only, capable of supplying full power and full rate to the control surfaces; the second supplying the utility services and also being kept as the standby system for the flight controls. Since it would only be the standby flight control system, it need not be capable of supplying full power to the flight controls, but only sufficient power to adequately manoeuvre the aircraft and control it to a safe landing.

Another possible combination is to use two separate systems for flight control operation, and a third for utility service actuation. This means three separate hydraulic systems. At a first glance this would appear to be less desirable than the two systems previously discussed.

However, if the two flight control systems are carefully combined, they can be made of the same power output level, and each need only have half the power output of the previously mentioned normal flight control system. This means smaller output pumps, smaller diameter lines and smaller servo valves.

Figure 1 shows this layout of hydraulic systems for flight control actuation.

Each system contains two hydraulic pumps – one driven from each engine. The surface actuators are of the tandem type, with a common output piston rod containing two pistons. Each half of the actuator is supplied with high pressure oil through its own servo control valve; the valves for each half of the actuator are machined out of a common spool and contained in a common housing. Pressure and return lines from each of the separate flight control hydraulic systems are connected to each half of the control valve.

There are two common points in this actuator. Common point means a place where system interconnection could occur.

The first is where the piston rod passes from one cylinder chamber to the other – careful seal design is required here to prevent system interconnection in the case of a single seal failure. In detail, each cyclinder has

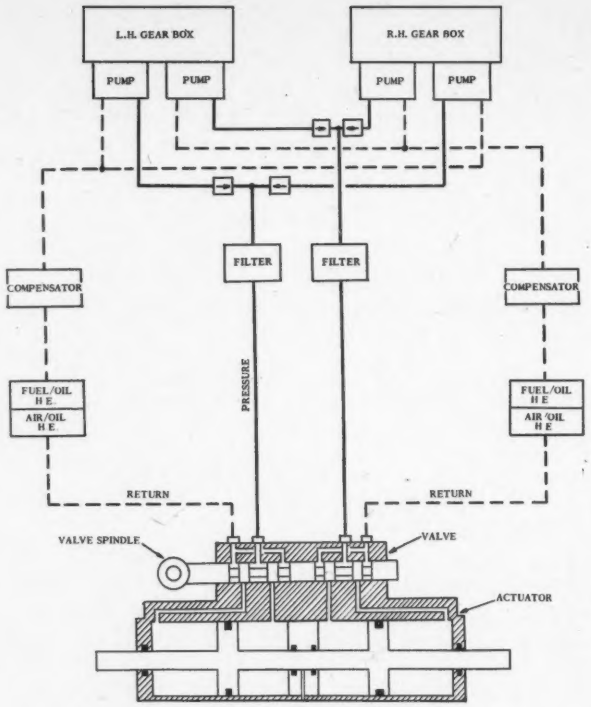

Figure 1

its own piston rod seal in the bulkhead, with the space between the seals vented to atmosphere.

The other common point is at the control valves; since the valve spools are machined out of a common piece of steel, interconnection could be possible. There are no dynamic rubber seals anywhere on these valves. This is necessary to keep the valve operating forces to an absolute minimum.

If a pressure differential does exist at any time across the lapped lands between the halves of the control valve, there will be a slight leakage from one system to another; but the leakage will be of the order of a few drops per hour and will have no noticeable effect on system operation or compensator level.

Considering the actuator as a complete unit, it is also obvious that a structural failure of the body of the cylinder in a critical location could cause a loss of fluid from both flight control hydraulic systems. It is therefore necessary to ensure that careful attention is paid to the detail design of the unit, and that adequate strength and fatigue testing is carried out.

Each flight control hydraulic system is sized so that when operating together they provide full power at full rate to the flight control surfaces.

In the case of a single *system* failure approximately full rate is obtainable with half maximum hinge moment. In the case of single *engine* failure full hinge moment is available at approximately half maximum rate.

While a tandem actuator is longer than a single one, capable of providing similar power output, it is only about half the diameter, and where these actuators have to fit in the trailing edge of thin supersonic wing sections, the diameter is the more critical dimension.

Having decided upon this system layout for the flight controls, the power required for the utility services can be obtained by the most efficient method, since it no longer has to tie in with the flight controls. Most efficient means from a combined weight, performance, cost, reliability and safety point of view.

The utility system

The duty cycle of the remaining controls is intermittent, but in the case of the landing gear, speed brakes, missile extension and nose wheel steering mechanism fast actuation is necessary.

The design objective for the operating time of these controls is in the order of:

(1) Landing gear – 5 seconds retraction time
(2) Speed brakes – 3 seconds open and shut
(3) Missile extension – 1½ seconds
(4) Nose wheel steering – 2 seconds to full caster

The size and weight of actuators required to move these controls, using electric power, would be excessive; plus the fact that it would not be possible to obtain operating times of the order required.

The advantages of pneumatics or hydraulics are not immediately apparent, as in the case of the flight controls, so a more extensive analysis of both pneumatic and hydraulic systems capable of operating these controls must be carried out.

Analysis of pneumatic operation of the utility system

To make the control actuators of a size small enough to be installed in the wings and fuselage a minimum useful pressure of 3,000 psi is required; to provide adequate storage capacity of high pressure air with as small a storage bottle as possible, a 5,000 psi compressor level was chosen, to be supplied with engine bleed air at a nominal 50 psi.

The next phase in the analysis is to determine the size of actuators to provide the necessary output force to operate the controls throughout their complete cycles.

This is done by obtaining aerodynamic force curves plotted against control displacement for various parts of the flight envelope. In the case of the landing gear and armament, curves are also plotted of actuator force due to aircraft g loading against control displacement. These forces are then combined with the loads due to aerodynamic forces and miscellaneous forces due to friction etc, to obtain a complete picture of the actuator output requirements.

The design of these actuators is further complicated by the speed with which they have to extend and retract. Each main landing gear weighs approximately 1,000 lb, and high kinetic energy is developed by retracting this mass through 90° in two to three seconds.

A large portion of this energy has to be removed by suitably designed snubbers or dampers before the gear strikes its stops.

From the armament system viewpoint, the missile, being extended into the air stream prior to firing, is highly susceptible to large accelerations, so that similar dampers are required in the armament actuators.

Providing adequate damping at the power level required is difficult with pneumatics and it is necessary to use snubbers that dissipate energy by forcing a liquid through varying sized orifices to bring the terminal

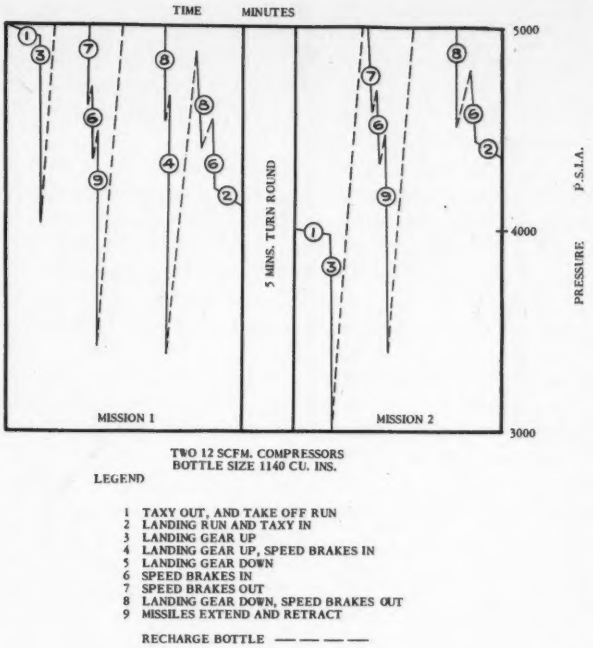

Figure 2

velocity to a satisfactory level, for example, about one to two inches per second.

We are now in a position to adapt the power requirements to the aircraft duty cycle using compressor capacity and storage bottle volume as the variable parameters. Figure 2 shows the pneumatic duty cycle using two 12 scfm 5,000 psi compressors and a 1,200 cu inch storage bottle.

Other combinations were analyzed, including one 16 scfm compressor and a 2,000 cu inch bottle, and two 16 scfm compressors with an 880 cu inch bottle. The weight estimate for the system using two 12 scfm compressors is given in Table 1.

Analysis of a comparable hydraulic system

The loading curves for the various control actuators have already been established, so the first job is to define the power system to supply these actuators.

Since an hydraulic system is not dependent on stored energy for its operation, the choice of system operating pressure can be higher than for a corresponding pneumatic system. If we consider the previously discussed pneumatic system, we must keep the pressure high enough to make the actuators small, but in doing this we have a bottle capacity below 3,000 psi which is of no use.

An operating pressure of 3,000 psi looks tempting, since this is a standard system pressure laid down in MIL-H-5440. This means that approved equipment can be bought "off the shelf", providing the system temperature can be kept to a 165°F maximum; even if it is necessary to operate at a higher temperature, certain equipment qualification tests could be waived on the basis of similarity.

An appreciable amount of theoretical work has been carried out to determine the optimum system pressure

for an hydraulic system. One such report "Theoretical Investigation of Optimum Pressures in Aircraft Hydraulic Systems", WADC Technical Report 54-189, Vol. I, by Conrad H. Cooke, shows that pressure to be in the order of 4,000 psi, which is in agreement with other papers on the subject.

As mentioned previously, it was necessary to use 4,000 psi as the operating pressure for the flight control system, which means that units will have to be developed capable of operating at this pressure. Some of these units can also be used for the utility hydraulic system.

The rate of control surface movement for the flight control system dictates the use of pumps delivering 10 USgpm at 1,500 rpm. Pumps of this capacity can operate the utility services at the required rates without resorting to the stored energy of hydraulic accumulators, so the same pump was chosen to power all the hydraulic systems on the aircraft.

Calculations similar to those performed on the pneumatic system were carried out to determine line sizes etc, and a weight analysis of a 4,000 psi hydraulic system was drawn up as shown in Table 1. It is only fair to mention that the hydraulic system weight is based on an actual system, whereas the pneumatics was based on preliminary schemes only.

Choice of power source for utility system

The weight analysis contained in Table 1 covers one aspect which must be considered when choosing the power source for the utility system. Other considerations, as previously mentioned, are: reliability, complexity, performance, safety and cost.

Reliability and complexity are closely related since the reliability of a unit decreases as the number of its component parts increases.

Since hydraulic systems have been used far more extensively in the past than pneumatic systems, they have built up a large backlog of experience, which, fed back into design concepts, results in the excellent reliability we expect from modern equipment and systems.

Performance is a factor for which a lower limit must be set during system design; equipment must be designed as close as possible to this limit, unless bonus performance can be obtained without an increase in cost or weight.

The question of relative safety between hydraulic and pneumatic systems hinges mainly about two points: the inflammability of hydraulic fluid, and the explosive decompression of the stored energy in the pneumatic system. It is easy to make an hydraulic system leak proof, and pneumatic storage containers and equipment can be made strong enough that explosions are unlikely. It seems to be six of one and half a dozen of the other.

While cost cannot be ignored in the design of a military aircraft, it is usually a deciding factor only when all the other factors show no significant advantage, one way or the other.

It is difficult to give a direct comparison of relative cost for hydraulic and pneumatic equipment. It depends to a large extent on whether the equipment is "standard" or "non-standard".

Considering the example used, the 5,000 psi maximum pneumatic pressure was higher than standard equipment

TABLE 1

Equipment	Weight lb	
	Hydraulics	Pneumatics
POWER CIRCUIT		
Pumps	31.4	
Accumulators	19.0	
Compensator	24.0	
Heat exchangers	4.0	
Filter and element	8.7	
Pressure regulating valve	10.2	
Pressure control valve	9.1	
Bypass control valve	3.00	
Thermal control valve	3.54	
Pressure switch	0.82	0.82
Compressors (engine driven)		62.5
Storage bottle		31.0
Water separator and dehydrator		12.5
Pre coolers		9.5
Valve – ground charging	0.75	0.75
Compressor (top-up – electric motor driven)		25.00
	114.51	142.07
LANDING GEAR		
Actuators — Main gear	31.5	35.0
— Nose gear	11.0	13.0
— Main door	28.0	31.0
— Nose door	5.28	6.0
— Uplocks	7.60	9.5
Valve — Selector(s)	6.3	6.5
— Sequence	6.65	6.65
— Transfer	1.57	1.57
— Restrictor	2.71	2.71
— Shuttle	3.00	3.00
— Reducing		2.25
Emergency storage bottle	11.7	11.7
Emergency selector valve	0.6	0.6
Snubbers — Main gear		20.0
— Nose gear		8.0
	115.91	157.48
SPEED BRAKE		
Actuators	21.54	25.0
Selector valve	3.6	3.0
Reducing valve		0.6
	25.14	28.6
WHEEL BRAKES		
Brake valve	11.2	11.2
Shuttle valve	0.5	0.5
Accumulator	15.3	
Storage bottle		10.5
Reducing valve		0.60
	27.0	22.80
ARMAMENT		
Selector valves	80.0	75.0
Actuators	144.4	158.0
Snubbers		51.0
Reducing valves		1.8
	224.4	285.8
NOSEWHEEL STEERING		
Valve – selector	2.00	1.00
Actuator and steering valve	18.00	25.50
Reducing valve		0.60
	20.00	27.10
MISCELLANEOUS		
Pipe, fittings, fairleads, etc.	344.0	107.0
Fluid	136.7	44.0
	480.7	151.0
TOTALS	1007.66	814.85
DIFFERENCE	+192.0	

in use; the 4,000 psi hydraulic pressure is also non-standard. However, a lot of development is required to produce the 4,000 psi hydraulic equipment for the flight control system, and since some of this equipment can be used on the utility system, a certain amount of saving can be realized.

When the aircraft engines are shut down, the engine driven pneumatic compressors stop and the storage bottles will be pressurized at some pressure below 5,000 psi (about 4,200 psi from Figure 2). Normal valve leakage will drop this pressure slowly as the aircraft sits on the ground; a top-up electric motor driven compressor is included in the system to bring it up to the 5,000 psi level before flight. The drawback to this is that it requires electrical power on the ground.

When all these factors are considered, the choice between pneumatics and hydraulics becomes very difficult since this intermittent type of operation is where the high pressure pneumatic system shows at its best. The choice was finally made in favour of hydraulics, on the basis of greater reliability and lower cost.

VARIOUS PROBLEMS IN HYDRAULIC SYSTEMS DESIGN

The choice of fluid

The state of the art in hydraulic fluids has progressed rapidly over the past few years, urged on by the increasing demand of supersonic aircraft and missiles for high temperature hydraulic fluids.

However, in 1953 the only fluid that had had even reasonably extensive service use was the standard red hydraulic fluid to MIL-O-5606 specification.

The decision was made to use this fluid and limit the system operating temperature to −65°F to 250°F with local "hot spots" to 275°F.

The choice of tube connections

At the time we had to decide upon a type of fitting assembly, i.e. flared or flareless; the flared fitting was the the standard USA type. However, great things were predicted of the flareless fitting, and we were given the impression that it would soon become the USA standard. To conform with what we believed was the trend, (and to be able to take advantage of the development work we hoped would be done in the USA on flareless fittings, used in environments similar to ours) we decided to use flareless fittings.

Since at this time little was known about the behaviour of flareless fittings in the temperature range −65° to 250°F, and in vibration spectrum we expected to work in, Avro embarked on an extensive development program in conjunction with the National Research Council in Ottawa.

From an operational viewpoint there are two points in favour of flareless fittings:

(1) Both Avro and USA manufacturers have found that flareless fittings seem to give a tighter system in practice; this could be due to the fact that they will seal with a less torque than flared fittings.

(2) That we can do away with the AN universal fitting screwing into the AND 10050 boss, using instead an inverted flareless fitting (Figure 3). The major problem with the AN universal fitting in service is that it requires careful assembly procedure or it will leak under pressure. On the CF-100 we had problems with this type of connection, due to incorrect assembly technique by both our own and service personnel.

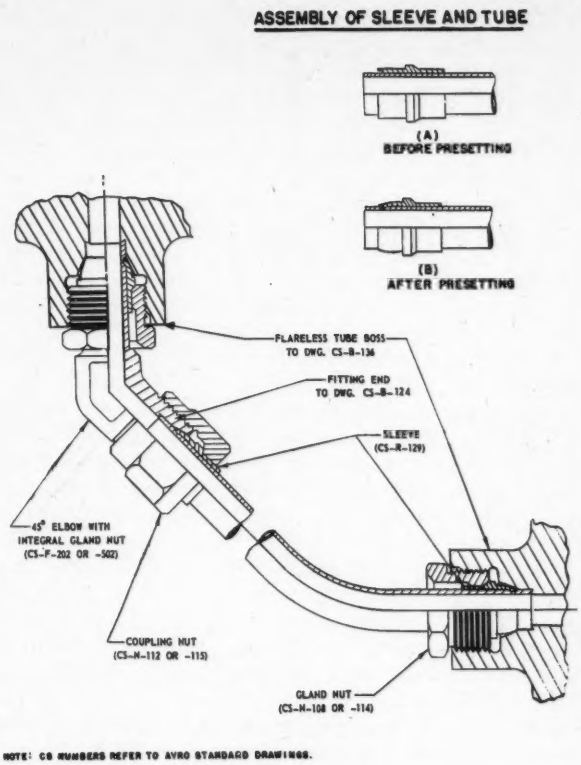

Figure 3
Flareless tube connections

The Avro flareless fitting design is different from the MS in the geometry of the sleeve, the method of "presetting" it on the tube, and the use of "Nylock" inserts in the nuts to prevent loosening under vibration.

The method of pre-setting is to swage the cutting edge of the sleeve into the tube in a pneumatically operated machine, while simultaneously a die swages down the tail of the sleeve into close contact with the tube. The pre-setting of standard MS flareless fittings is basically the same, except the tail of the sleeve is not swaged to the tube.

Initial test on assembly samples proved that at room temperature they will withstand 10,000,000 cycles of flexural stress (± 35,000 psi measured at a point on the tube's extreme fibre, opposite the end of the sleeve) with an internal fluid pressure of 5,000 psi. Some problems were encountered, however, at 275°F.

The original specimens had an endurance of from 700,000 to 2,400,000 cycles, at 275°F. A microscopic examination of these specimens suggested that their premature failure may have been due to an increase in fretting corrosion between the tube and the sleeve with the high temperature. Two attempts were made to cure this problem. Firstly, by substituting non-corrodible plating for the cadmium plating on the sleeve and, secondly, by putting layers of an inert compound, during assembly, between the sleeve and the steel tube. Neither of these approaches showed any consistent improvement.

The next approach was, firstly, to decrease the amount of tail swaging and, secondly, to lower the core strength of the sleeve. Again no appreciable improvement was obtained. All this testing was carried out on ½″ o/d × 0.035″ tube to MIL-T-6845, ⅛ hard.

While we were carrying out this program to produce an improvement in endurance without changing our basic production assembly method, the NRC were carrying out a development program to determine ultimate fatigue limitations of both flared and flareless fittings. As a result of this program they developed a technique of assembling flared and flareless fittings by expanding the end of the tube until its outside diameter becomes an interference fit (+ 0.002″ – 0.0005″ on ½″ tube), with the particular sleeve to be used.

Our main problem then was to develop a satisfactory production assembly based on this technique. It seems that at long last we have reached this goal, after many attempts, but the final proof will have to wait till more specimens assembled in this manner have been tested.

Another problem associated with tubing installation is the supply of high pressure to hydraulic equipment moving with respect to the structure. In lower pressure, lower temperature systems, this can be accomplished by using flexible hoses. There was no hose available to operate in our environmental conditions, so we had to resort to some other device.

On the services which are only operated a few times in flight, i.e. landing gear, speed brakes etc, swivel fittings and rigid tubing are used; on the flight control actuators, which are moving continuously and for the most part over small angular movements, we decided to use spring pipes – that is rigid piping with the installation designed to allow for flexure as the actuator extends and retracts. The bodies of the actuators are pinned to the structure and they pivot over about 1° for full control surface movement. The original design called for 4130 steel tubing with brazed end fittings.

An extensive test program was started on each specific installation design, combining pressure impulses with representative pipe flexing. We hadn't progressed far in the program before we hit trouble.

On one set of pipes, the test rig, which closely resembled the aircraft installation, was designed to oscillate the pipes vertically through an angle of ± ½°, at a frequency of 10 cps; oscillate in a horizontal plane, through + 0 – 1° at a frequency of 22 cps; and oscillate longitudinally along the jack, to which the moving ends of the pipes were attached, through + 0.06″ – 0.04″ at a frequency of 11 cps.

At the same time the pipes were subjected to pressure pulses at approximately 0.8 cps; each pulse consisting of 70% time at 4,000 psi constant pressure; followed by a fall to 600 psi and subsequent rise to 6,000 psi on an approximately sinusoidal wave form. The entire test rig was maintained in an ambient temperature of between 225° and 250°F. The test requirement was to complete 2½ million cycles of the 10 cps motion, with the other motions and pulses occurring in the proportions shown above.

Two types of failures occurred after only about 2% to 4% of the total number of cycles required. The most trouble experienced was due to failure at the junction of the pipe and the brazed end fittings. The other type of failure was in the tube, due to surface irregularities and corrosion on the inside of the tube.

It was then decided to try using stainless steel pipe with flareless fittings. These pipes lasted about 40% to 50% of the required number of cycles, the failures occurring at the bends or loops in the pipes. The next step was to test standard 180° bend samples to determine the effect of ovality on fatigue strength. This program is still in progress to achieve the original design aim of 2½ million cycles.

The main problem associated with this type of program is to determine representative duty cycles with respect to motion and fluid pressures. These can only be accurately determined by extensive rig and aircraft flight testing.

Reservoir design

One of the limitations that exists when using MIL-O-5606 at reasonably high temperatures (250°F) is that every attempt must be made to keep air out of contact with the fluid of the system. The bleeding of air from the operating portions, and the detail design of components to prevent air from coming out of solution, are basic design aims of any hydraulic system – no matter what temperatures or pressure it operates at. But if air is kept in intimate contact with hot hydraulic fluid in the system reservoir, the fluid will combine with the oxygen in the air to form sludge, which if allowed to remain in the system will greatly reduce its performance.

Once this sludge has formed, it is only possible to remove it by draining and flushing the system, and by removing the majority of components for bench cleaning. To overcome this problem an airless reservoir, or compensator, (Figure 4) which supplies pressurized oil

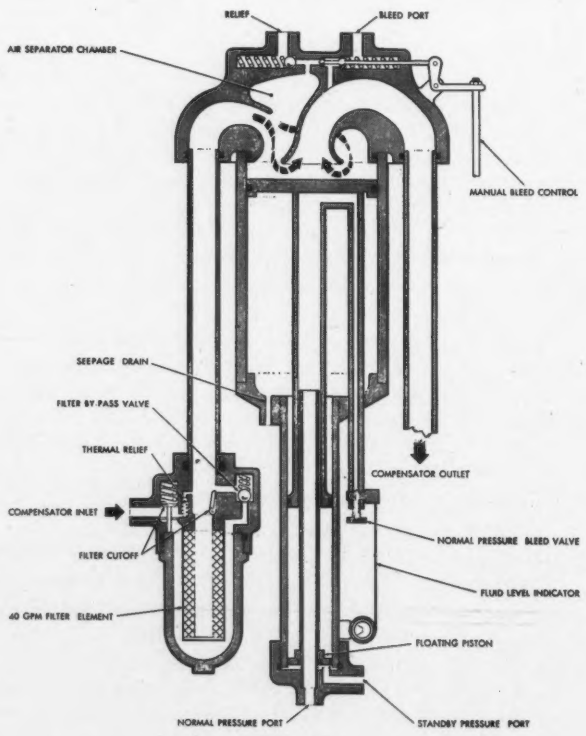

Figure 4
Flying control hydraulic system—compensator

to the pump inlets under all operating conditions and aircraft attitudes, was designed.

Fluid under 1,500 psi pressure is supplied to the small piston on the compensator; the relative areas of the large and small pistons reduces this to a nominal 90 psi for return line pressurization. This is commonly referred to as the "bootstrap" technique.

The compensators are filled with reference to system temperature. Each system has a strap-on type thermometer, placed in an easily readable location, and on a pipe where an "average" system temperature is approximated.

A measuring device, attached to the compensator piston, travels in and out as the piston moves. The temperature, read off of this device at the reference level, is compared with the system temperature read off of the thermometer; if necessary fluid is added or removed from the system to make these two temperature readings coincide.

The head of the compensator is designed to remove most of the air in the system by inducing a vortex at the oil inlet; the separated air goes to the top of the unit where it can be bled off by manual operation of a bleed valve.

The initial design had an O ring sealing on the large piston, but operating experience showed that this was prone to rolling and two or three cases of spiral O ring failure occurred. Test work was then started to compare the relative merits of O rings and quad rings for this application. (The quad rings have a four lobed cross section.) To date, the quad rings have shown superior characteristics.

The temperature problem

A preliminary analysis of the heat input to the hydraulic systems and the heat loss from the systems by conduction, convection and radiation, indicated that to keep the maximum average system temperature to 250°F some form of cooling would be required.

As previously mentioned, there are three separate hydraulic systems, each getting power from two 10 gpm, at 1,500 rpm and 4,000 psi, variable delivery pumps.

Considering first the operation of each of the flight control hydraulic systems with respect to heat generation, heat will be generated by pressure drop in the supply lines, filter etc, to and from the pumps and control valves, and by the pumps and valves themselves.

The average power output from each flight control hydraulic system is in the order of 20 hp, which at a supply pressure of 4,000 psi amounts to a flow of 8.5 USgpm. This flow is made up of leakage flow in the stability augmentation servos, and an average rate of control surface movement during flight.

There is also a case drain flow out of the pump case back to system return in the order of 1 to 1½ gpm; this remains sensibly constant, regardless of pump output flow. Experience has proved that this case drain flow must be put back into the system as close to the compensator as possible, to prevent excessive pressure build up in the pump casing. This means that the heat taken away from the pumps by the case drains has to be removed by warming up some fluid in another part of the system.

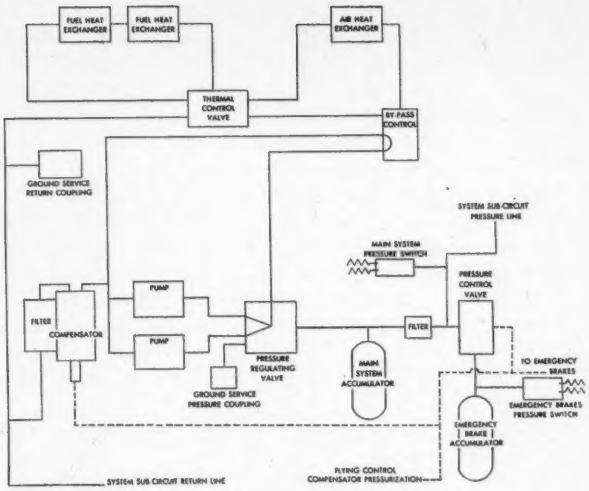

Figure 5
Power circuit block diagram

The final heat exchanger array chosen was to pass all the fluid returning from the control units through, first, an air/oil heat exchanger, then through a fuel/oil heat exchanger. This fluid would then mix with the hot fluid coming from the pump case drain lines, resulting in an acceptable fluid temperature level.

The air/oil heat exchanger has an integral relief or bypass valve. This valve is designed to bypass the fluid past the heat exchanger core if the pressure drop in the core exceeds a given value; this has the effect of allowing the system to warm up rapidly after a cold start. It also has the added effect of preventing excessive return line pressures at these temperatures. A very small flow is bled through the core when the bypass valve is open to flush the cold fluid out of the core. The fuel/oil heat exchanger has an integral thermal control valve. This valve is set to open at 200°F to allow the hot oil to pass through the core and be cooled by the fuel flow.

This valve is necessary to prevent excessive heat from being put into the fuel when the hydraulic system is still at an acceptable temperature level, since the fuel is used to cool other systems on the aircraft.

The cooling of the utility hydraulic system presents a different problem. The selector valves for the utility services have no quiescent flow and the system's average flow is effectively zero, except when a particular service is being operated.

This means that all the heat generated by the two pumps in the system has to be carried away by the case drain lines. Calculations showed that this flow was not enough to keep the system temperature to a 250°F maximum, even when passing through the two heat exchangers; some other form of power supply was needed.

Various combinations of dual pressurized and electrically depressurized pumps were considered, but the best and lightest solution turned out to be the use of constant displacement pumps with a pressure unloading valve. This type of system has the advantage of full pump output flow to carry away the heat generated. Figure 5 shows the heat exchanger array used on the utility hydraulic system. When system pressure builds up to

the unloading valve cut out pressure of 4,250 psi, the full pump output is bypassed to return. To keep the heat exchanger size to a minimum, only a maximum of 11 gpm of fluid is passed through the heat exchangers; this is mixed with the hot fluid, which is bypassed by the heat exchangers, resulting in an overall lowering of fluid temperature.

There are two control valves in this circuit – a bypass control and a thermal control valve. When the temperature of the fluid leaving the unloading valve is below 90°F, all the flow is fed directly into the pump inlets; as the temperature increases above 90°, the bypass control valve starts to divert the flow through the air/oil heat exchanger. If the temperature increases to 215°F, the fluid starts to pass through the fuel heat exchanger as well.

When the pumps are turning at maximum speed the full pump flow passing through this circuit causes a back pressure at the pumps of about 250 psi. This results in a steady power loss of about 6 hp maximum or 245 Btu/min. This value will decrease, of course, as the pump speed decreases. Compare this with a variable delivery pump, which puts out a steady 300 Btu/min heat loss per pump, or 600 Btu/min per system, coupled with the low flow available to remove this heat.

Filter arrangement

An hydraulic system will continue to operate satisfactorily only if it is kept clean; the word *clean* is, however, a purely relative one, and obviously the degree of cleanliness required depends on the vulnerability of the equipment in the system to contamination. The cleaner the system has to be kept the more money it will cost, both in filtering equipment and preventive maintenance procedures.

The units most susceptible to contamination are those with small operating clearances and highly polished working surfaces. The servo controls in a flight control hydraulic system are perfect examples of this and are also critical pieces of equipment from a safety point of view. A lot of thought has been put into the pattern of system filtration, both by industry and the A6 Committee of the SAE.

An extensive program is being carried out to determine, amongst other things, the best method of measuring the amount of contamination present in a system, and also the degree of filtration necessary for use with high performance servo valves. It was decided to use 10 micron nominal filtration on the hydraulic systems. All the equipment specifications include the requirement that the equipment must operate satisfactorily under these conditions. The filter elements used are nominal 10 micron and will retain 98% of all particles 10 microns or larger in their two largest dimensions, with an absolute cut off of 25 microns. They are made of stainless steel woven mesh.

The basic system filtration consists of two high capacity filters, one which filters all the fluid output from the pumps, and the other which filters all the fluid before it is returned to the system compensator. Small additional filters have been placed upstream of some servo valves, which are relatively remote from the main pressure filter; these small filters also filter to a 10 micron nominal level.

The systems, as previously mentioned, are closed with respect to the atmosphere and they are pressure filled with fluid that has been filtered to a 5 micron level. These features help to keep out dirt at two points in the system where it is often introduced, i.e. during system filling or topping up and at aspirated reservoirs. A continuous contamination program is being carried out to check the degree of contamination present in the systems, at regular time intervals, to get a picture of the effectiveness of the filter array and to determine the amount and type of contamination generated by the system.

The most common place for dirt to be generated is in a pumping device. A few years ago, Chance-Vought and Vickers Hydraulics of Detroit devised a method of checking pump acceptability, from a dirt producing angle, prior to installation. This is called the "patch" method and has been discussed in detail in other papers.

Experience has shown that pumps which are liable to produce an excessive amount of dirt in service, and possibly fail prematurely, can be picked out by careful testing and examination prior to installation in an aircraft. Samples taken from filters at the inlet, outlet and case drain ports of the pump during run in are compared with standard samples or patches obtained from a clean pump. If these samples show a heavier contamination level than the standard samples, the pump is rejected. This is important from a system standpoint to ensure that these units will not add excessive contaminant to the system.

THE FUTURE OF HYDRAULICS

There are two basic parameters in an hydraulic system – pressure and temperature.

The parameters which directly affect system operation, such as fluid viscosity, bulk modulus, lubricity etc, depend to a large extent on the temperature limits of the system, since these determine the type of fluid used in the system.

Pressure

MIL-H-5440B (25 May, 1955) "Design, Installation and Tests of Aircraft Hydraulic Systems" defines the standard system pressures. The maximum pressure defined is 3,000 psi.

Theoretical work which has been carried out in England and the USA over the past few years shows that the optimum pressure is in the region of 4,000 psi.

A preliminary military specification has already been prepared for a standard 4,000 psi system.

The need to go to even higher pressures still exists where high horsepower is required and equipment installation space is limited despite the increase in weight involved.

Temperature

MIL-H-5440B divides the systems into three types according to their temperature operating limits.

Type 1 has temperature limits of –65° to 160°F, Type 2 has temperature limits of –65° to 275°F, and Type 3 of –65° to 400°F.

The lower temperature limit for all three types has remained at –65°F. The Type 1 system has a temperature spread of 230°F between the upper and lower

limits, and the Type 3 system has a 465°F spread, or twice as much as the Type 1 system. The larger this temperature spread is, the more difficult it becomes to make the system and its components operate satisfactorily at both the upper and lower limits.

As higher aircraft speeds are reached it will no longer be practical to provide cooling adequate to keep the average fluid temperature down to a 400°F maximum.

The next logical series of upper temperature limits would possibly be 550°F, 700°F and 1,000°F.

The temperature problem can be broken down into two parts, the fluid temperature in the system and the ambient air temperature surrounding the system components. If the ambient air temperature is higher than the fluid temperature, then heat will be added to the system from the surrounding air as well as from internal generation. When the ambient air temperature is higher than the allowable maximum fluid temperature, the fluid must be continuously circulated to prevent stagnant fluid from becoming overheated.

Fluids

In assessing the suitability of a fluid for a particular application, it is necessary to consider its chemical and physical properties.

The properties that affect the operation of an hydraulic system are:

(1) Viscosity – temperature properties
(2) Lubrication and wear
(3) Vapour pressure
(4) Shear stability
(5) Oxidation – corrosion stability
(6) Hydrolytic stability
(7) Effect on metals and elastometers
(8) Bulk modulus
(9) Specific heat and thermal conductivity
(10) Toxicity
(11) Co-efficient of thermal expansion
(12) Spontaneous ignition temperature
(13) Flash point
(14) Pour point
(15) Density
(16) Foaming characteristics and gas solubility

Type 1 systems use fluid to MIL-O-5606 almost exclusively with some civilian operators using Skydrol, a "less inflammable" hydraulic fluid.

Type 2 systems can be made to operate satisfactorily with the petroleum based fluid MIL-O-5606, provided an airless system is used.

The military specification for Type 3 hydraulic fluid is MIL-H-8446A. To date only one fluid has been approved to this specification; that is Oronite's MLO-8515, a blend of 85% disiloxane (a special type of silicate ester) and 15% dioctyl sebacate, with added viscosity index improver, and oxidation and hydrolysis inhibitives.

The group of organic silicate ester base fluids appear to be satisfactory for Type 3 systems although they suffer from poor lubricity.

In the temperature levels above 400°F several fluids show promise at temperatures up to the 550° to 700°F level. Oronite MLO-8200, which is 100% disiloxane with inhibitors, is good for limited time at 550°F when used with metal seals or heat resistant elastometers. It has better high temperature viscosity and stability characteristics than MLO-8515, but is not satisfactory for use with standard rubber seals due to its "low swell" property.

General Electric's Versilube 81717, a chlorophenyl-silicone fluid, appears to be suitable between –65° and 575°F with a special lubricity additive.

Dow Cornings QF-6-7009 is stable between –25° and 550°F and for short periods up to 700°F.

New hydrocarbon compounds are being developed which appear to be as good as any of the synthetic fluids in this temperature range.

Between 700°F and 900°F the polyphenyl ethers seem to be the best, but with present fluid knowledge system operation will be limited to at the most a few hours before the fluids break down.

For 1,000°F operation various types of synthetic and hydrocarbon blends are being considered. Liquid metals, such as sodium-potassium eutectics, are being investigated for use in fluid power transmission systems. Since they solidify at about 15°F, this will obviously be the lower temperature limit of the system. However, whatever material is used as the basic stock for fluids at this temperature level, it is a foregone conclusion that the lower temperature limit will not be below 0°F and will most likely be around the 10°F mark. The liquid metals present corrosion, spontaneous ignition and other minor problems which still have to be overcome.

Seals

A parallel development program is also being carried out on the problem of high temperature sealing.

For the Type 2 systems packing to MIL-P-5516 is good up to 225°F and MIL-P-18017 to 275°F. No seal we have yet tested has been satisfactory over the complete range of –65° to 275°F although Viton "A" to MIL-P-25732 looks very good.

Viton "A" and Neoprene "W" have proven satisfactory in laboratory tests up to 400°F with Neoprene "W" having better low temperature characteristics.

For temperatures of 550°F and above the metallic seal appears to be the only answer. As in all seal design, the major problem is associated with dynamic sealing, and especially dynamic sealing to prevent external fluid leakage. Apart from the obvious fact that external leakage will eventually deplete the supply of fluid, the temperatures at which the systems are operating are often above the flash point of the fluid and external leakage will present a serious fire hazard.

The main parameters that have to be considered in the design of metallic seals are:

(1) fluids
(2) materials, and
(3) surface finish.

High temperature fluids suffer generally from poor lubricating properties; this makes the problem of seal wear and life critical. Associated with this is the fact that many types of metallic seals require close tolerance machining and installation for proper functioning and fluid contamination must be carefully considered. The selection of proper materials for the seal and rubbing surface is essential. Galling cannot be allowed. Best results have been obtained when piston rods have been

nitrided, flame plated or treated with a dry lubricant, but the effect of high temperature fluids on methods of hardening or plating of the rods must be watched.

As far as surface finish is concerned, the more important factor is the finish of the piston rod rather than the seal itself. Vertical lapping to 3-5 microinches rms gives good results with seal finishes of 16 microinches rms plus a honing operation. It is essential that the surfaces are not scratched, either during fabrication or in use, as this will produce early failure in endurance testing.

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R.Ae.S. — C.A.I. — I.A.S.

SEVENTH

ANGLO-AMERICAN CONFERENCE

5th to 16th October, 1959

NEW YORK — CLEVELAND — BUFFALO — TORONTO

C.A.I. LOG

SECRETARY'S LETTER

HALIFAX-DARTMOUTH AND BADDECK

EARLY in May I visited Nova Scotia, though I cannot claim that my primary purpose was a visit to the Halifax-Dartmouth Branch. However, I did see several members of the Branch on my way through en route to and from Baddeck, and I wish I could have spent more time with them.

As it was, I was met at Halifax airport by CDR E. B. Morris, Chairman of the 50th Anniversary Committee, and travelled with him to Baddeck by car. He had been having difficulty in getting work started on the monument that the Institute is erecting in celebration of the anniversary and, with the unveiling due in June, he suggested that we should visit the place together to settle the various issues on the spot.

We had an entertaining couple of days, which included a tour around some of the lesser known by-ways of Cape Breton in search of stone. This was a problem which we had not expected; surely the good people of Baddeck would know where their "local stone" was to be found – in a quarry or some such place. But not a bit of it. The local stonemason, who was our guide, knew only of a likely headland, and when we got there we discovered that he was looking for tumbled-down barns, which might or might not have stone basements. After much tramping over sodden fields in the rain we found a suitable derelict; we even succeeded in tracing the owner and made a deal with him for the purchase of his basement walls.

This was typical of our adventures and the "advancement of the art, science and engineering relating to aeronautics" never assumed a less familiar form. We left Baddeck with a fervent prayer that our efforts had not been in vain and that things would at last start to roll and keep on rolling. As I write the progress reports are pretty good.

JOURNAL MATERIAL

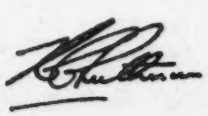
We continue to be showered with both praise and criticism of the papers appearing in the Journal. We bow gratefully to the one and will do what we can to satisfy the other. The Executive Committee of the Council and the Publications Committee have given a good deal of thought to the subject and I believe that their plans should enhance the interest of the papers for the average member – whoever he may be. But, of course, we inevitably come back to the point that papers do not write themselves and it is not the job of Headquarters staff to write them, even if we could.

If our members could only be persuaded to write about their own work, they would be surprised what interest it would arouse. Most of us who are interested in our jobs – and we should not be members of the Institute if we were not – are inclined to forget that other people might be interested too, particularly in any practical experiences which are rather out of the ordinary. Neat solutions to everyday problems; problems that seem insoluble and on which we should like advice; the application of new techniques to whatever our jobs happen to be – all these are material suitable for the Journal if only people would put them down on paper and send them in. Literary skill is of no great importance; we can do a little rehashing here, if it is necessary and if the article is not too long. The important thing is the idea; and we all have ideas sometimes.

It would be helpful too, if members could let me know what they would like in the way of subject matter. If we publish a few suggestions, they might start people writing about them. The other day somebody suggested "Salvage"; now has anyone got anything to write about Salvage? And has anyone any other subjects to suggest?

LIST OF MEMBERS

As an economy measure, it has been decided that the List of Members will not be reissued in 1959. However we hope to publish particulars of the Institute, Branch and Section Committees in a future issue of the Journal.



BRANCHES

NEWS

Calgary

Reported by H. E. Hampshire

April Meeting

The first Annual General Meeting of the Calgary Branch was held on the 23rd April at the Officers' Mess, RCAF Station Lincoln Park. An unexpected heavy snowfall with resultant hazardous road conditions resulted in an attendance of only 25 members and guests.

Various reports of Branch activity for the 1958-59 season were made by the Secretary, Treasurer, Program Committee Chairman and the Councillor. The Branch Chairman, Mr. W. A. B. Saunders, summed up the reports submitted and thanked the executive and all members for their loyal support.

Mr. A. R. G. Leckie, Chairman of the Branch Student Awards Committee, announced the winner of the CAI bronze medallion as Mr. W. Mitchell, a student at the Institute of Technology and Art, for his presentation of a technical paper entitled "Non-Destructive Inspection of Aircraft Material and Components". A token presentation was made to Mr. Mitchell by Mr. Leckie.

The results of the election of Branch officers for the 1959-60 season were announced.

On behalf of the Branch, Mr. W. E. Jamison thanked the outgoing executive for all their efforts during the season and called for continued support from every member in order that the Branch may grow and continue to contribute to the overall effectiveness of the Institute.

At the conclusion of the business session three interesting films were shown. The evening was rounded off by a general social get together.

Halifax-Dartmouth

Reported by Lt. J. A. Turner

April Meeting

The Third Annual General Meeting of the Branch was held in the cinema of the Chief Petty Officers' Mess, HMCS Shearwater, on Wednesday, 15th April. A total of 19 members and guests were present.

The highlight of the meeting was the presentation of the Student Award by the Chairman, CPO R. L. S. Sabourin, to Mr. J. M. Mullins. Mr. Mullins, who is currently attending the Nova Scotia Technical College, was selected as the first winner of this award because of his outstanding grades in Fluid Dynamics.

The Chairman then read the Annual Report of the Executive Committee which showed that the Branch was firmly consolidated in every respect with the exception of the provision of suitable speakers. He expressed the hope that, during the forthcoming year, several tour speakers could be provided in order to alleviate this situation. The Financial Report was then presented and adopted as read.

The business portion of the meeting was concluded with reports from the Fiftieth Anniversary and Banquet Committees.

After a brief intermission, films on "Cylinder Combustion" and "Flutter" were shown.

Vancouver

Reported by G. W. T. Roper

April Meeting

The 5th Annual Branch Meeting was held on the 11th April at the Canyon Gardens, with approximately 200 members and guests in attendance. The Chairman opened the meeting by welcoming the members and guests. He then expedited a brief but important business session.

The adoption of the minutes of the previous meeting was moved, seconded and approved.

Mr. R. J. McWilliams reviewed briefly the activities of the past year and predicted the future growth of the Branch. He also explained that no additional names had been received by the Nominating Committee for the new executive and that the adoption of the Nominating Committee report would elect the officers as slated in the report. The adoption of each of the reports of the Chairman, Secretary, Treasurer, Councillors, Program Committee, Membership Committee and Nominating Committee was moved, seconded and approved.

Mr. R. N. McCollum moved that a debt of honour in the amount of $50 be paid to Headquarters. This motion was seconded and approved unanimously. Mr. McCollum thanked Mr. McWilliams for his efforts on behalf of the Branch, not only in the past year as Chairman but from its initial conception. Appreciation was expressed by all.

The Chairman thanked his executive for their help in the past year and introduced the incoming executive. The meeting was then turned over to the new Chairman, who spoke briefly.

There being no further business, the meeting adjourned and all enjoyed a social program of dining and dancing.

Montreal

Reported by I. S. Macdonald

May Meeting

The Branch Chairman, W/C C. R. Thompson, opened the meeting by stating that this was the final dinner meeting of the current season. He then asked the Treasurer, Secretary and Chairman of the Student Activity Committee to cover the final reports.

Mr. E. C. Norsworthy thanked the outgoing executive for their past year's efforts.

The Chairman then addressed his final remarks and thanked his own executive committee for their help during the year. He also outlined the main events that took place during the year, namely the Golden Anniversary Banquet, the Golf Tournament of last summer and the success in the area of Student activity.

W/C Thompson also answered several questions regarding the finances of the Montreal Branch and with the help of Mr. Boyd outlined the status of Headquarters' finances.

Finally the Chairman introduced the new executive and turned the meeting over to the new Chairman for the coming year, Mr. R. F. O. Smith.

Mr. Smith thanked the old executive for their work and mentioned that there

would be an informal meeting during the summer to provide continuity to the work of the new executive.

Approximately 30 members attended the dinner. After the formalities a short movie on the Convair 880 was enjoyed by all.

Edmonton

April Meeting

Reported by P. F. Leigh-Mossley

The ninth meeting of the season was held at 700 Wing RCAF Association at 8.00 pm on the 8th April.

Mr. C. C. Young, Branch Chairman, welcomed 15 members and 3 guests and called on the Secretary to read the minutes of the last meeting. It was then moved by Mr. C. Arnold and seconded by Mr. J. Bristow that the minutes be adopted as read.

The Chairman then read the names of the new Executive Committee for 1959-60.

Mr. Young stated that we had had an offer from Douglas Aircraft Co. Inc. of Santa Monica of a speaker for the 12th May. The speaker originally booked for that date had agreed to postpone his talk until the 27th May.

The Chairman then asked for any suggestions or criticisms from the floor on the past year's meetings. It was agreed that the programme for the past year was of a very high standard. The question of getting more members to attend the meetings was thoroughly aired and it was suggested the meetings be held on a different day each month.

The meeting was then adjourned, proposed by F/O A. J. Robinson and seconded by Mr. C. Arnold.

A very excellent buffet supper was then enjoyed by all concerned.

May Meeting

Reported by P. F. Leigh-Mossley and S/L J. E. Moran

The tenth meeting of the season was held at 700 Wing RCAF Association at 8.00 pm on the 12th May.

In the absence of our Branch Chairman, the chair was taken by F/L K. Weinstein who welcomed 17 members and 11 guests.

The Secretary read the minutes of the ninth meeting. It was then moved by Mr. C. Arnold and seconded by S/L J. E. Moran that the minutes be adopted as read.

The Chairman then asked S/L Moran to introduce the guest speaker for the evening, Dr. T. J. Schultz, Assistant Chief of Acoustics, Douglas Aircraft Co., Santa Monica, Calif.

Dr. Schultz presented a paper on "The Interior Acoustics of the DC-8 Aircraft". Since the movie "The First Flight of the DC-8" had been shown at the November meeting, this was a timely topic and, it turned out, a fascinating one.

The speaker briefly reviewed the basic differences in the noise problem between the propeller and the jet aircraft. In the former, the principal noise source was the propeller, followed closely in importance by the exhaust; the effects of these sources were felt mainly in the fuselage areas adjacent to the propeller-engine combination. These noises were mostly of low fundamental frequencies not in the speech interference spectrum, and so the acoustical designer could cope with them by rather easy methods. The noise of the jet transport, however, was generated in the jet wake and increased slightly along the length of the fuselage; it was an unpredictable mixture of all frequencies, including those of speech. At higher speeds even the jet noise was not as important as the noise generated by the turbulent boundary layer along most of the length of the fuselage.

To attenuate the noise entering the fuselage, the designer has the usual three sound transmission factors to consider: first, the mass of the barrier (the fuselage wall) which could not be increased except with a very unattractive weight penalty; second, the number of leaks in the surface which could not be reduced easily because of air conditioning requirements and the increased problems of production tolerance; third, the absorption characteristics of the interior surface. The last method was the only one which offered practical scope.

Acoustically induced, metal fatigue was also a much more serious problem in the jet aircraft. In the propeller driven aircraft, only panels or members with natural frequencies near the fundamental frequency of the propeller in cruise conditions and located near the areas of maximum sound intensity would likely offer problems. In trouble areas, the panel's natural frequency could be easily changed. Because all frequencies were present in the jet wake or in the burbling boundary layer, all panels were exposed at times to their natural frequencies, and so fatigue might occur anywhere and a change of gauge or section would not necessarily cure the problem.

Dr. Schultz then outlined the actual methods employed in aircraft to combat the noise problem and emphasized the approach in the DC-8 and some of the intended approaches in the DC-9.

In conclusion, the speaker described the design considerations for the public address system of the DC-8 and how Douglas was proposing to meet them.

This was a paper on a subject with which most of the Branch members were unfamiliar. Its enthusiastic reception indicated the clarity and quality of the speaker's presentation. The speaker was thanked by F/L Weinstein.

Cold Lake

Reported by WO2 J. W. Day

May Meeting

The last meeting of the season of the Cold Lake Branch was held on the 14th May. The attendance numbered 10 members.

The talk scheduled by F/L Tarzwell was cancelled and will be given in the fall when the Branch commences the new season's lectures.

F/L L. S. Lumsdaine, in handing over to the next Executive Committee, recommended that for the next season the Programme Committee schedule the meetings to suit the many functions that take place at this station and to suit the members.

Mr. J. B. Panton asked for a vote of thanks for the retiring Committee and for the excellent and extra effort during their term of office.

F/L Lumsdaine then asked the members to a social evening at his home which was much enjoyed.

MEMBERS

NEWS

I. A. Gray, F.C.A.I., was recently appointed to the Canadian Airworthiness Council, representing scheduled airlines.

R. D. Hiscocks, F.C.A.I., was recently appointed to the Canadian Airworthiness Council, representing aircraft constructors.

H. S. Rees, F.C.A.I., was recently appointed Chairman of the Canadian Airworthiness Council.

R. D. Richmond, F.C.A.I., has been appointed Vice-President in charge of the newly formed Missiles and Systems Division of Canadair Ltd.

F. W. Austin, A.F.C.A.I., will be retiring in June from his position with the Air Registration Board.

G. L. Best, A.F.C.A.I., formerly Asst. General Manager of Northwest Industries Ltd., has been appointed General Manager of B.C. Air Lines Ltd., which was recently purchased by Northwest Industries Ltd.

N. J. G. Hill, M.C.A.I., has been appointed Director and General Sales Manager of Air Trainers Link Ltd., Aylesbury, England.

M. W. MacLeod, A.F.C.A.I., has retired from the position of Lines Development Engineer with Trans-Canada Air Lines.

A. M. Scott, M.C.A.I., has been appointed Chief Engineer of Rolls-Royce of Canada Ltd., Aero Division.

J. A. van der Bliek, M.C.A.I., will be leaving Ottawa in June to take up a position with ARO, Inc., Tullahoma, Tenn., in their new Hypervelocity Branch.

E. L. Bunnell, M.C.A.I., has been appointed Sales and Contracts Manager, Aircraft, for Bristol Aero-Industries Ltd. (Winnipeg Division).

J. A. Pool, M.C.A.I., was recently appointed to the Canadian Air Worthiness Council, representing non-scheduled airlines.

J. G. Mitchell, Technical Member, has been appointed Contracts Administrator for Genaire Limited.

ADMISSIONS

At a meeting of the Admissions Committee, held on the 22nd April, 1959, the following were admitted to the grades shown.

Associate Fellow

A/V/M J. A. Easton, RCAF, Air Member for Technical Services, AFHQ, Ottawa, Ont.

G/C W. M. Diggle (on transfer from Member)

LCDR (E) (AE) E. Gosh (on transfer from Member)

Member

F/O J. E. Dardier, RCAF, Flying Instructor, RCAF Stn. Cold Lake, Alta.: *Officers' Mess, RCAF Stn. Cold Lake, Alta.*

Lt. (E) (AE) N. B. Davis, (on transfer from Technical Member)

E. Davison (on transfer from Technical Member)

A. C. R. Schropfer (on transfer from Technical Member)

Technical Member

N. P. Christenson, Engine Test Inspector, Orenda Engines Ltd., Malton, Ont.: *28 Rectory Rd., Weston, Toronto 15, Ont.*

G. A. Clement, Service Engineer–Editorial, Canadair Ltd., Montreal, P.Q.: *4792 Rachel St., Parish of St. Genevieve, P.Q.*

R. J. Cronk (on transfer from Junior Member)

B. R. Dudley, Assistant to Accessories Supervisor, Rolls-Royce of Canada Ltd., Montreal, P.Q.: *100 Forest Gdns., Point Claire 33, P.Q.*

W. H. Friend (on transfer from Student)

D. R. Madill (on transfer from Student)

F/O G. P. Stevens (on transfer from Student)

F/O D. H. S. Wright (on transfer from Student)

Junior Member

R. Hay-Roe (on transfer from Student)

Student

C. R. Cousins, Sir George Williams College, Montreal, P.Q.: *Hudson Heights, P.Q.*

E. I. Mackie, University of British Columbia, Vancouver, B.C.: *1550-16th Ave., Burnaby 3, B.C.*

G. A. S. Owen, Fort Camp, University of British Columbia, Vancouver, B.C.

P. B. Sauriol, Provincial Institute of Technology and Art, Calgary, Alta.: *1417 - 7th Ave. N.W., Calgary, Alta.*

D. G. Strang, University of British Columbia, Vancouver, B.C.: *5519 President's Row, Vancouver 8, B.C.*

E. W. Watchorn, University of British Columbia, Vancouver, B.C.: *2643 W. 13th Ave., Vancouver 8, B.C.*

R. Wirsig, St. Stephen's College, University of Alberta, Edmonton, Alta.

Associate

J. R. Audette, First Officer, Canadian Pacific Airlines, Edmonton, Alta.

* * *

At a meeting of the Admissions Committee, held on the 19th March, 1959, the following were admitted to the grades shown.

Associate Fellow

A. J. Bachmeier, Section Head, Gas Dynamics, National Research Council, Ottawa, Ont.: *Box 33, R.R. 1, Ottawa, Ont.*

W. H. Brokenshire, Metallurgical Engineer, Development and Research Division, The International Nickel Company Ltd., 55 Yonge St., Toronto, Ont.

Member

G. E. Dawson, Instructor, Ryerson Institute of Technology, Toronto, Ont.: *Clearview Heights, King City, Ont.*

D. Ellington, Engineer, Special Weapons Engineering, Canadair Ltd., Montreal, P.Q.: *300 - 14th Ave. N., St. Eustache sur le Lac, P.Q.*

H. Etingin, Section Head, Electrical Engr. and Liaison Engr., Canadair Ltd., Montreal, P.Q.: *1440 Poirier St., Apt. 107, St. Laurent, P.Q.*

R. A. le Faucheur, Technical Writer, Canadair Ltd., Montreal, P.Q.: *11881 Michel Sarrazin, Apt. 201, Montreal 9, P.Q.*

A. B. Oxley, Reliability Coordinator, Special Weapons, Canadair Ltd., Plant 4, P.O. Box 6087, Montreal, P.Q.

G. P. Peterson, Controls Engineer, Canadian Pratt & Whitney Aircraft Co. Ltd., P.O. Box 10, Longueuil, Montreal 33, P.Q.

J. W. Tanney, Technical Officer, National Research Council, Ottawa, Ont.: *585 O'Connor St., Apt. 4, Ottawa, Ont.*

Dr. N. B. G. Taylor, Superintendent, Physiology & Biochemistry Wing, Defence Research Medical Laboratories, Toronto, Ont.: *King High Drive, Concord, Ont.*

Technical Member

G. T. C. Bedwell, Aero Repair Shop Supervisor, Sperry Gyroscope Co. of Canada Ltd., Vancouver, B.C.: *1432 Compston Cresc., R.R. 2, Ladner, B.C.*

G. A. Bird, Production Planner, Trans-Canada Air Lines, Montreal, P.Q.: *695 Laurentian Blvd., No. 9, St. Laurent, Montreal 9, P.Q.*

G. W. Doran, Design Engineer, Canadair Ltd., Montreal, P.Q.: *11955 Grenet St., Apt. 4, Montreal, P.Q.*

R. M. Korchynski, Mechanical Engineer, Trans-Canada Air Lines, Winnipeg, Man.: *402 Martin Ave., Winnipeg 5, Man.*

G. R. Salter, Thermodynamics Engineer, Canadair Ltd., Montreal, P.Q.: *Box 100, Ile Bellevue, Brucy, P.Q.*

F/O A. C. Trupp, RCAF, Aircraft Servicing Officer, RCAF Stn. Winnipeg, Man.: *181 Ethelbert St., Ste. 3, Winnipeg 10, Man.*

J. F. Whitmore, Technical Author, Vickers Armstrong Aircraft Ltd., Weybridge, Surrey, England: *12 Chumleigh Walk, Surbiton, Surrey, England.*

A. Ziolkowski, Process Planner, De Havilland Aircraft of Canada Ltd., Downsview, Ont.: *112 Sheldon Ave., New Market, Ont.*

Junior Member

P. S. Moller, Thermodynamics Engineer, Canadair Ltd., Montreal, P.Q.: *1625 Grenet, Apt. 15, Ville St. Laurent, Montreal, P.Q.*

Student

D. C. Guglielmin, Provincial Institute of Technology and Art, Calgary, Alta.: *1124 - 20 Ave. N.W., Calgary, Alta.*

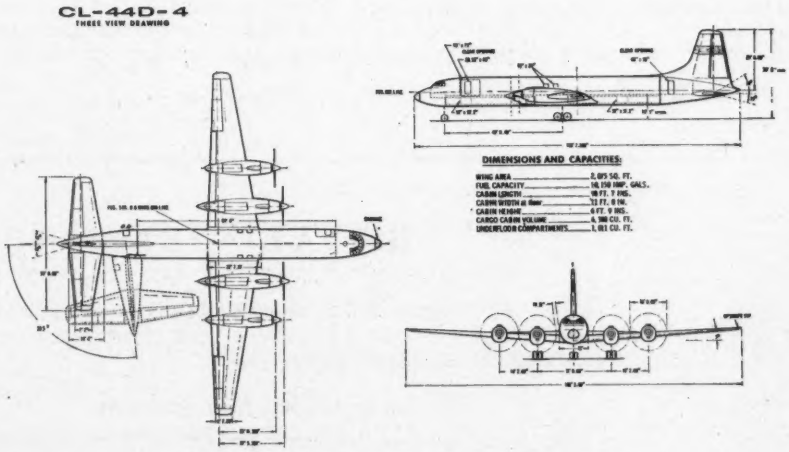
SUSTAINING MEMBERS

NEWS

Canadair Ltd. has begun deliveries to the USA of the sets of wings and ailerons for the Bomarc which are to be produced under the new "defence production sharing" arrangements. The components are produced under subcontract with the Boeing Airplane Company. The missiles concerned are for the USAF; it is stated that none of the components contracted for so far at Canadair will go into the Bomarc ground-to-air missiles planned for Canadian installations.

At the time the contract was announced in Washington on the 23rd February, it was interpreted as a test of Canadian production facilities, even though the monetary value was small. In the intervening 10 weeks before the first deliveries were made, an accelerated production program was carried out at Canadair in order to meet the exacting delivery schedule. In that time shipments of tools from Seattle had to be installed at Canadair, production crews trained and assembly areas organized. Moreover, permanent sources of supply had to be established and many new manufacturing processes introduced.

Canadair Convair 540 (Napier Eland engines)



Canadair CL-44D-4 (Rolls-Royce Tyne engines)

Canadair aircraft are also in the news. A tour of South America by a Canadair-Convair 540 (two Napier-Eland engines) began at the end of April for demonstration to airline operators, owners of executive aircraft and military authorities. Another 540 is now touring Canada and the USA and a third will set out on an European tour in June.

In addition, Canadair has received orders for the CL-44 long-range transport from the two largest all-cargo airlines in the USA. (This aircraft with its swing-tail feature was described in the March issue of the Journal.) The Flying Tiger Line Inc., with headquarters in Burbank, Calif., has ordered ten of the aircraft and Seaboard & Western Airlines Inc., of New York, has ordered five. The total cost involved for the 15 aircraft, including spares, is about $70 million. In addition, each airline has taken an option on five more of these aeroplanes.

The aircraft is powered by four Rolls-Royce Tyne 12 engines. It has a cruising speed of over 400 mph and will carry a payload of 65,000 lb across the North Atlantic. Its usable volume is approximately 7,400 cu ft and the swing-tail feature gives great advantages in loading and unloading. With a speed some 30% higher and a direct operating cost 40% less per ton mile than the most economical commercial cargo planes now in operation, the CL-44 represents a major advance and will permit tariffs approaching the cost of surface transport.

Deliveries will begin in 1961.

Northwest Industries Ltd. has expanded its operations to British Columbia by the purchase of B.C. Air Lines Ltd. of Vancouver. B.C. Air Lines, based at Vancouver Airport for its operations serving Vancouver Island, Powell River and Queen Charlotte Islands, was acquired from Paul Tak and Associates and the sale includes the entire fleet of 21 amphibious and float aircraft together with a new hangar housing the servicing shops and facilities of the Company.

Under the new ownership, it is planned to continue all the present airline operations of the Company while the shop and repair facilities will now be operated by Northwest Industries. This will enable B.C. Air Lines to offer, through their connection with Northwest Industries, repair and overhaul facilities for the largest commercial and private aircraft operators in British Columbia. The Edmonton plant is one of Canada's four largest plants engaged in the repair, modification and overhaul of aircraft for the RCAF and is the only wholly Canadian-owned plant engaged in such work. The Edmonton plant employs 600 persons at their 40 acre plantsite located on Edmonton's Municipal Airport. In addition to the repair of aircraft and aircraft radio equipment, it has one of the finest instrument laboratories in western Canada.

Railway & Power Engineering Corporation Ltd. and Canadian Flight Equipment Cobourg Ltd., in association with Servomechanisms (Canada) Ltd. and Tamper Ltd., have formed a team for the Canadian manufacture of equipment designed by Lear Inc.

Railway & Power Engineering will perform the overall management function and will be responsible for coordination of the activities of the manufacturing associates as well as for the conformance of products with conditions of contract and specification.

Canadian Flight Equipment will produce mechanical components. This company has had wide experience in this particular field, notably with cartridge actuators and time release mechanisms, and is well equipped to handle those Lear components requiring precision machining. The electronic and electric components of the Lear equipment will be manufactured by Servomechanisms (Canada) and Tamper respectively.

The purpose of the plan is to produce Lear equipment – such as automatic flight control systems, flight reference systems, gyros, stable platforms, servos and actuators – with the highest possible Canadian content and, by using facilities already in existence in the country, to do so at prices comparable with those prevailing in the USA.

Canadian Applied Research Ltd. and Phoenix Engineered Products Ltd. have now been merged by A. V. Roe Canada Ltd. into a single corporate entity which will continue in the name of Canadian Applied Research Ltd. The operations previously carried out by Phoenix will now become the Repair and Overhaul Division of CARL. In connection with this expansion, CARL's Manufacturing, Sales and Service, and Finance Divisions, together with the administrative offices, have moved from 1500 O'Connor Dr. to 750 Lawrence Ave. West. The Engineering Division will remain at 1500 O'Connor Dr.

ROYAL VISIT 1959

H.R.H. The Prince Philip has graciously consented to address engineers and scientists at a luncheon at the Royal York Hotel, Toronto, at noon on Monday, June 29th. Groups participating are:

CANADIAN AERONAUTICAL INSTITUTE
CANADIAN COUNCIL OF PROFESSIONAL ENGINEERS
CANADIAN INSTITUTE OF MINING AND METALLURGY
CHEMICAL INSTITUTE OF CANADA
ENGINEERING INSTITUTE OF CANADA

Applications for tickets should be made immediately to:

Engineers and Scientists Committee
P.O. Box 62,
Postal Station Q,
Toronto 7, Ont.

The ticket allotment will be one to a person and the applicant must indicate his association with one of the organizations participating. Tickets are non-transferable.

A cheque in the amount of $5.00, payable at par in Toronto, must accompany each individual application. Cheques should be made payable to the Engineers and Scientists Committee.

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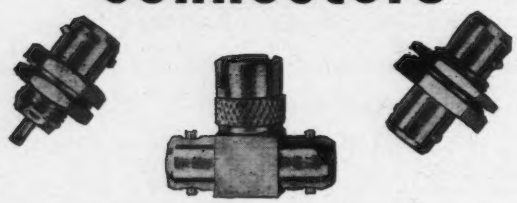

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ST. CATHARINES, ONT.

INSTRUMENTATION ENGINEER

NATIONAL AERONAUTICAL ESTABLISHMENT

National Research Council
Ottawa, Ontario

National Aeronautical Establishment, Ottawa, requires a graduate engineer to take charge of the Instrumentation Group on the new Five Foot Mach 4.5 Wind Tunnel. He would be responsible for the design and development of instrumentation for wind tunnel models and for development of data handling and processing equipment and wind tunnel controls. He would also supervise the professional and technical personnel of the instrumentation group and take charge of the instrumentation laboratory. He must be familiar with the applications of strain gauges, electrical transducers, automatic electronic and hydraulic controls, and analogue and digital data processing systems. He must be capable of an original approach to a large variety of advanced instrumentation and data handling problems.

Candidates must be graduates with honours from a recognized university in Electrical Engineering or Physics (or equivalent) and must have several years of pertinent experience.

Initial salary up to $9,000.00 per annum is offered for this position depending on the qualifications and experience of the successful applicant.

Apply giving age and complete details of education and experience to the Employment Officer, National Research Council, Sussex Drive, Ottawa 2. Please quote file ME-551.

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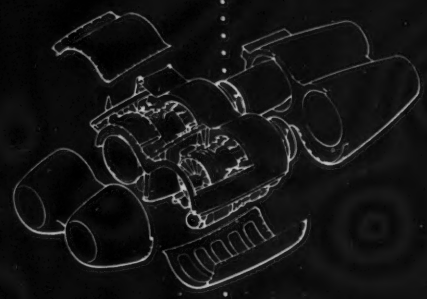
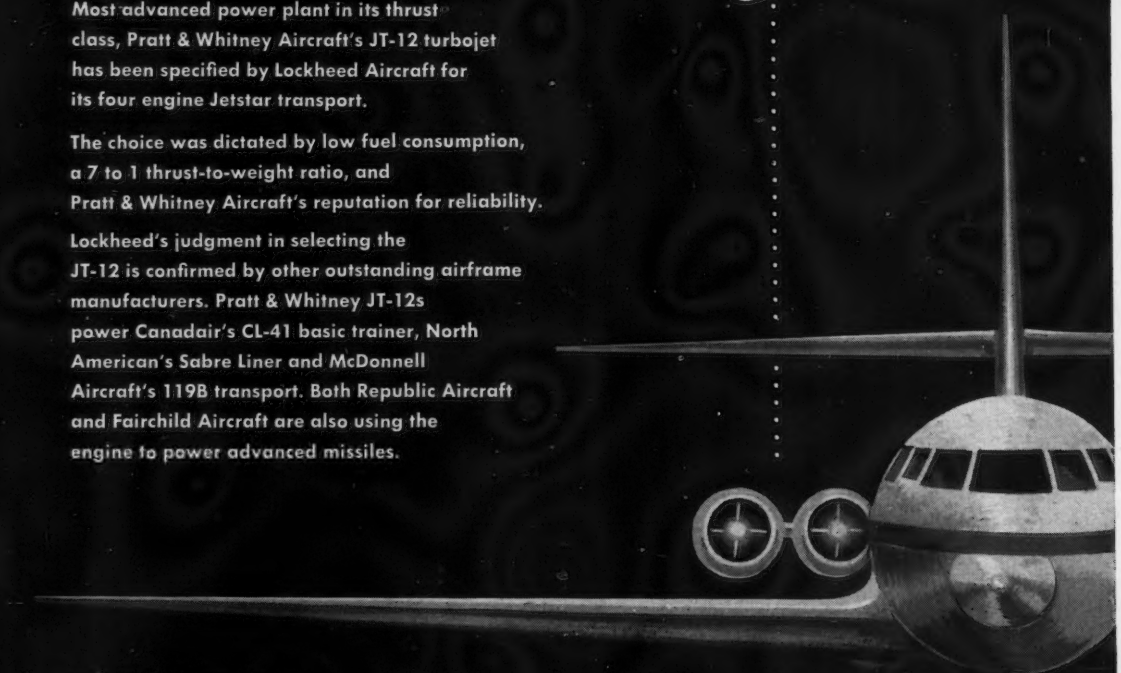


Most advanced power plant in its thrust class, Pratt & Whitney Aircraft's JT-12 turbojet has been specified by Lockheed Aircraft for its four engine Jetstar transport.

The choice was dictated by low fuel consumption, a 7 to 1 thrust-to-weight ratio, and Pratt & Whitney Aircraft's reputation for reliability.

Lockheed's judgment in selecting the JT-12 is confirmed by other outstanding airframe manufacturers. Pratt & Whitney JT-12s power Canadair's CL-41 basic trainer, North American's Sabre Liner and McDonnell Aircraft's 119B transport. Both Republic Aircraft and Fairchild Aircraft are also using the engine to power advanced missiles.



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- *Sikorsky Helicopters*
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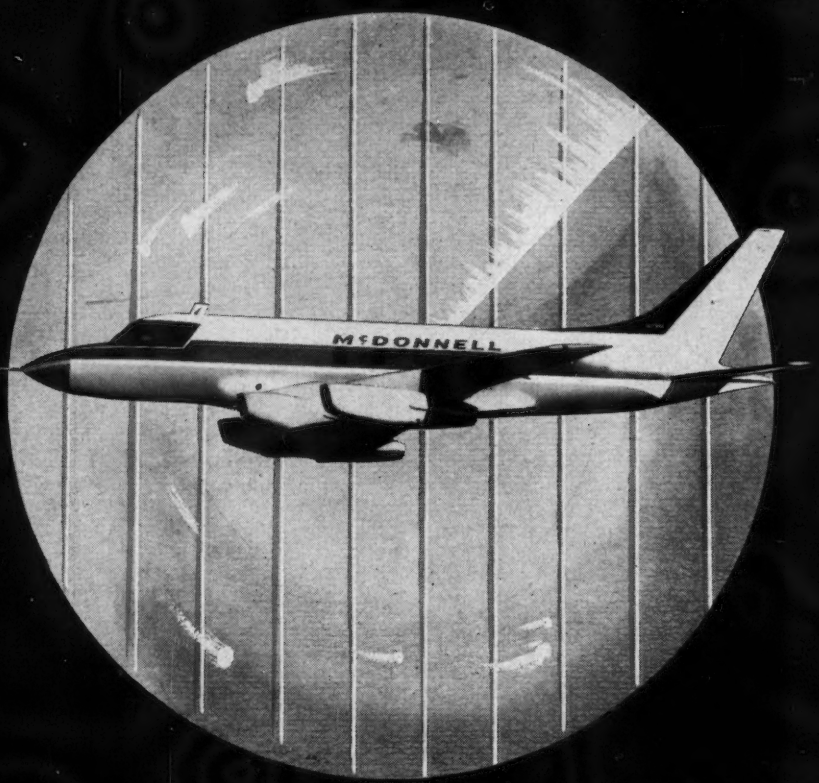
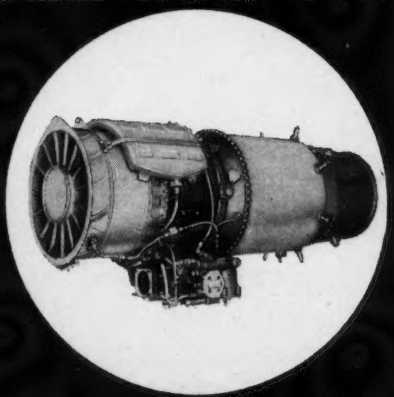
Longueuil – Quebec

McDONNELL SELECTS



PRATT & WHITNEY POWER FOR MODEL 119B TRANSPORT

Most advanced power plant in its thrust class, Pratt & Whitney Aircraft's JT-12 turbojet has been specified by McDonnell Aircraft for the Model 119B Transport. The choice, as in so many other cases, was dictated by low fuel consumption, a 7 to 1 thrust-to-weight ratio, and Pratt & Whitney Aircraft's reputation for reliability.



McDonnell's judgment in selecting the JT-12 is confirmed by other outstanding airframe manufacturers. Pratt & Whitney JT-12s power Canadair's CL-41 basic trainer, North American's Sabre Liner and Lockheed's Jetstar transport. Republic Aircraft and Fairchild Aircraft also use the engine to power advanced missiles.

Canadian Pratt & Whitney

Aircraft Company, Limited

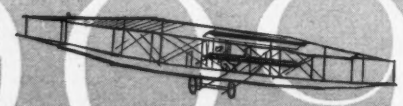
Longueuil, Quebec

Pratt & Whitney Engines · Sikorsky Helicopters · Hamilton Standard Products · Norden Electronics · Pesco Aircraft Accessories

SOME CANADIAN AIRCRAFT ENGINEERING MILESTONES

The first 50 years have seen many instances where Canadians have pioneered in world aviation. Here are but a few examples.

1909



First flight of The Silver Dart. Considering the absence of precedence, and the hazards which were instinctively overcome, this was indeed a major achievement.

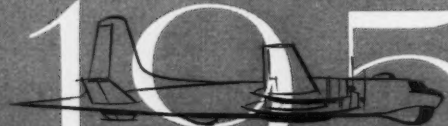
1924

First Canadian-designed aircraft, the Vickers Vedette, produced in Montreal. The Vedette did first arctic aerial photography, and was so generally useful that some were still in service at the beginning of World War II.

1942-45

Production of many important aircraft including plywood Mosquito bombers for World War II. Many hydraulic components for these Canadian-built aircraft were supplied by Jarry.

1957



Production of first CL-28, largest aircraft ever built in Canada. A complete re-design of Bristol Britannia by Canadair. Landing gear by Jarry.

1958

Production of first Avro CF-105, the world's fastest interceptor. World's first high temp/pressure hydraulic-powered controls, and nose gear by Jarry.

1959

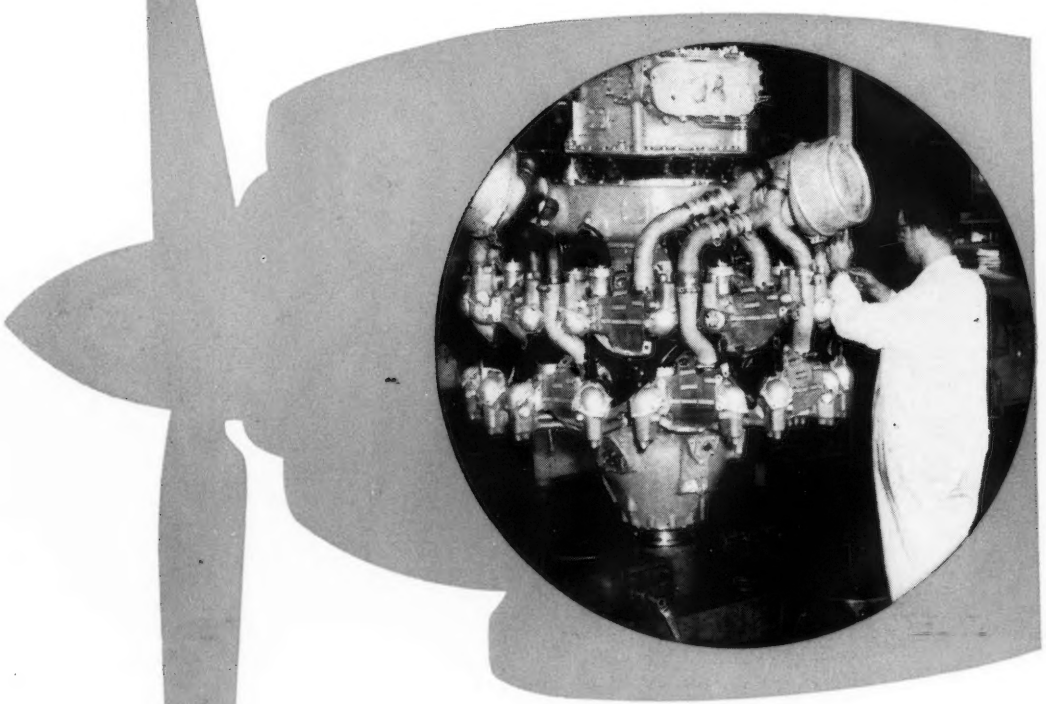
Delivery to U.S. Army of first De Havilland Caribou, Canadian-built, with outstanding STOL characteristics. The patented STOL Jarry Landing Gear causes enquiries from manufacturers throughout the world.

Jarry Hydraulics LIMITED

MONTREAL CANADA

2100 HRS. OVERHAUL LIFE

This is the overhaul life of the Wright 3350 Turbo-Compound engine overhauled by BRISTOL for a leading world airline.



IMPORTANT NOTE
FOR OPERATORS:

Avoid costly problems in the next two years during the switch from piston engines to prop-jets and jets. Let Bristol Aero-Industries overhaul your engines, reduce your costs and ensure completely overhauled and tested engines and components whenever you need them.

The Montreal Engine Division of Bristol Aero-Industries, with a complete overhaul service backed by the most modern facilities, has contributed in large measure to this remarkable achievement.

BRISTOL AERO-INDUSTRIES LIMITED
MONTREAL TORONTO WINNIPEG VANCOUVER

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2nd in a series on the aircraft and engine overhaul, servicing and manufacturing facilities of Bristol Aero-Industries Limited